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AIR FORCE MISSILE DEVELOPMENT CENTER
TECHNICAL REPORT

AIRCRAFT INERTIAL NAVIGATION SYSTEM

TEST PROGRAM INFORMATION

CENTRAL INERTIAL GUIDANCE TEST FACILITY

A F MISSILE DEVELOPMENT CENTER

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SEPTEMBER 1966

HOLLOMAN AIR FORCE BASE
NEW MEXICO

This document has been prepared for the information of those interested in the general test procedures followed by the Central Inertial Guidance Test Facility in testing aircraft inertial navigation systems. Included is a standardized test procedure to implement Department of Defense memorandum, dated 6 July 1965, issued by Dr. Harold Brown, Defense Director of Research and Engineering.

> ". . . the Central Inertial Guidance Test Facility is considered the DOD focal point for aircraft inertial navigator test and evaluation . . . the selection of aircraft inertial navigators for current and future avionics applications (will) be made from those navigators whose specified performance has been verified at CIGTF."

AIRCRAFT INERTIAL NAVIGATION SYSTEM

TEST PROGRAM INFORMATION

Prepared By

Operational Test Division
Central Inertial Guidance Test Facility
Air Force Missile Development Center

Foreword

The Central Inertial Guidance Test Facility (CIGTF) was established to provide an Air Force capability to test and evaluate the products of the inertial navigation and guidance industry. The goals to be achieved by the CIGTF are the following:

- Unbiased evaluation of components and systems, to provide data from which the customer can select the optimum equipment for a given mission application.
- Development of a single centralized test facility, to avoid the prohibitive costs of duplicated facilities.
- Standardization of tests, to provide common yardsticks for comparative evaluations.
- Competence in both personnel and equipment, to insure meaningful evaluations.

Originally established to provide test support for the development of early ballistic missile systems, the CIGTF has expanded its capability to cover the full spectrum of inertial navigation and guidance equipment. The development of advanced precision test facilities and the acquisition of a hard core of experienced personnel have produced an unequaled facility for the evaluation of missile, spacecraft, and aircraft systems and components. This growing competence has resulted in increased emphasis on the role of the CIGTF as a national focal point for navigation system testing. The test facility is available to the three services, NASA, FAA, and private industry.

Aircraft inertial navigation system tests began at the CIGTF in February 1964. Since that time, eight inertial navigation systems have undergone testing at the CIGTF.

Planning for an aircraft inertial navigator test program does not significantly differ in principle from planning for other test programs. Upon receipt of a test program request, the CIGTF will determine the resources required to perform the tests and prepare a test plan and program documentation for coordination with the customer. If necessary, a contribution to the customer's Technical Development Plan will be prepared.

Because nearly every test program has peculiar requirements, the earliest possible notice should be given to permit procurement of long lead-time items. The customer is expected to provide the test specimen

and necessary AGE. Arrangements for the provision of special test equipment will vary. The CIGTF will provide test bed aircraft, common test equipment, data reduction facilities, and test personnel. Familiarization of CIGTF personnel with the equipment is usually required at the contractor's plant. Contractor technical representation is desired at the test site.

Throughout the program the customer is encouraged to observe the tests. He is kept aware of significant occurrences through immediate informal reports. Upon completion of the test, the CIGTF prepares a complete engineering and data analysis report for distribution by the customer.

Requests for test support or further information regarding test programs, including the Standardized Test described in this document, should be directed to the Central Inertial Guidance Test Facility, addressed:

Hq AFMDC (MDS)
Holloman AFB New Mexico 88330

This technical report has been reviewed and is approved:

Robert B. Savage

ROBERT B. SAVAGE, Colonel, USAF
Director, Central Inertial Guidance Test Facility

ABSTRACT

The designation of the Central Inertial Guidance Test Facility (CIGTF) as the DoD focal point for aircraft inertial navigator test and evaluation required that a generalized test plan be written to govern all future tests. This document outlines such a Standardized Test, including test philosophy and objectives, the test approach and an outline of the test procedure. It provides the reader with an understanding of the CIGTF aircraft inertial navigator test capabilities, the types of test programs currently available, and the requirements necessary for an agency to enter systems in these programs. Six appendices, which cover areas such as analysis methods, laboratory testing, instrumentation, are included to provide the project test engineer with additional detailed information.

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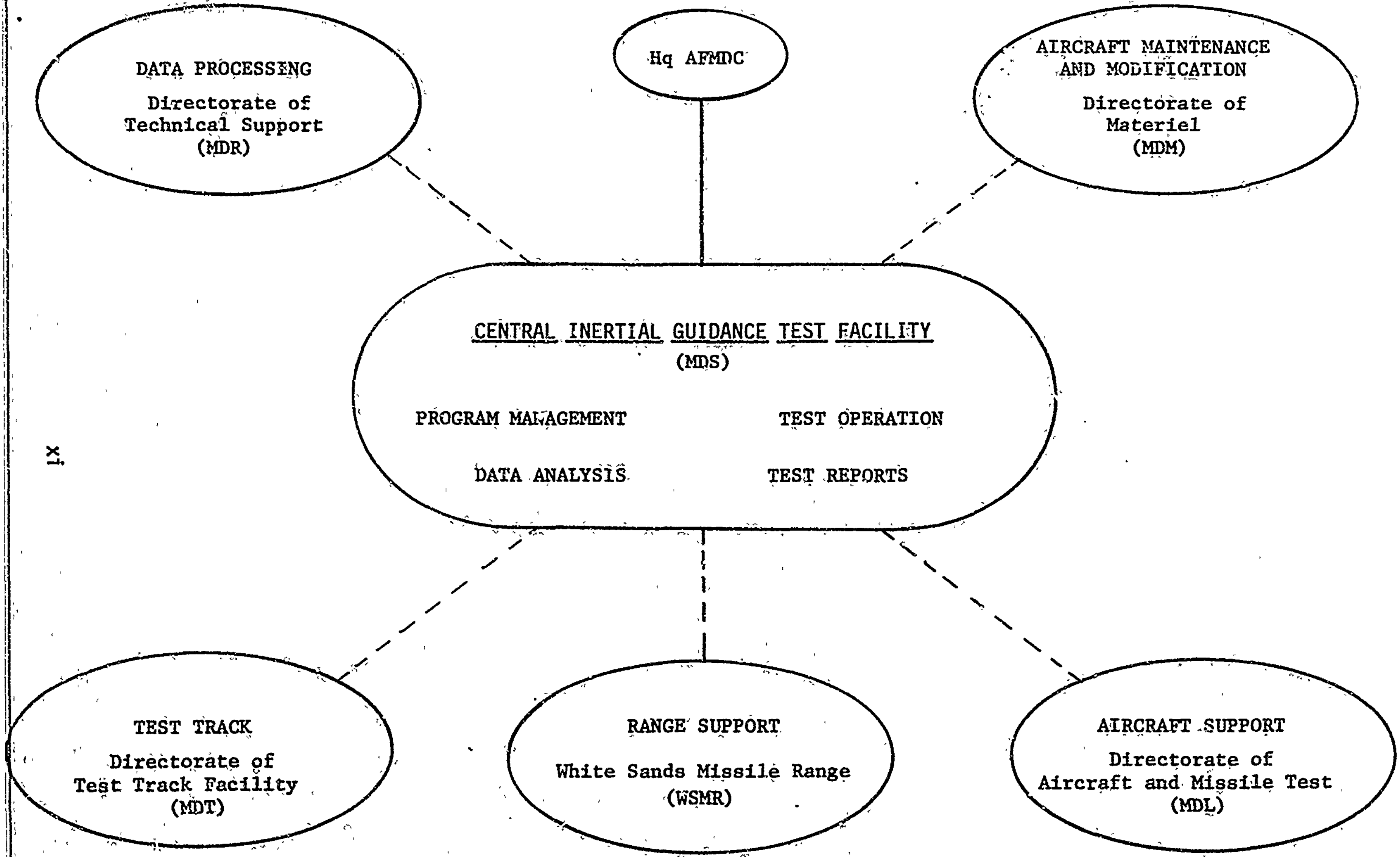
DEFINITIONS AND ABBREVIATIONS

Error Plot	A plot of a system indicated position (velocity) minus reference position (velocity) versus flight time.
Quick-Look Error Plot	A position error plot that is manually produced and can be made available within 24 hours after a test.
Reduced Error Plot	An error plot that is machine (computer) produced by matching FPS-16 radar tapes and system output tape.
FPS-16 Radar	A highly accurate tracking radar used for most inertial navigation test missions. When tracking an aircraft radar beacon, position accuracies attained are less than 100 feet. Velocity measurements are accurate to about 1 fps over the ranges flown.
Cinetheodolite	An optical tracking device for obtaining position and velocity information. Particularly useful in meeting strict position accuracy (5 - 10 ft) requirements and in obtaining precise measurement of instantaneous vehicle velocities (0.5 fps).
DOVAP	Doppler velocity and position tracking reference. A doppler space position tracking system providing highly accurate velocity measurements. Velocity accuracies attainable are on the order of 0.1 fps over the ranges flown.
Check Point	Accurately surveyed geodetic position used as a reference to determine a test vehicle's true position.
KTAS	Knots True Airspeed
MSL	Mean Sea Level
CIGTF	Central Inertial Guidance Test Facility
AFMDC	Air Force Missile Development Center Holloman Air Force Base, New Mexico
WSMR	White Sands Missile Range, New Mexico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(MDS)
PROGRAM MANAGEMENT
TEST OPERATION
DATA ANALYSIS
TEST REPORTS
TEST TRACK
Directorate of
Test Track Facility
(MDT)
RANGE SUPPORT
White Sands Missile Range
(WSMR)
AIRCRAFT SUPPORT
Directorate of
Aircraft and Missile Test
(MDL)

Figure 1

1. INTRODUCTION

1.1 The purpose of this document is to provide the reader with an understanding of the CIGTF test capabilities, the types of test programs currently available, and the requirements necessary for an agency to enter systems in these programs. The document has been designed to implement the intent of Department of Defense memorandum dated 6 July 1965.

1.2 Organization

1.2.1 The CIGTF with the support of other AFMDC agencies and WSMR provides the capability for complete test and performance evaluation of inertial navigation systems within the Department of Defense (see Figure 1). This provides not only unbiased performance evaluation, under conditions closely simulating an operational environment, but also results in greater economy than does contractor testing.

1.2.2 The CIGTF has overall program management responsibility for these tests. In addition to identifying resource requirements and preparing test plans and program documentation, the CIGTF performs laboratory and flight tests, accomplishes statistical analyses of test data and prepares engineering and analysis reports.

1.2.3 The Directorate of Technical Support operates an extensive analog and digital computation facility for the reduction of test data. The heart of this facility is a Control Data Corporation 3600 digital computer.

1.2.4 The Directorate of Test Track Facility operates the 35,000 foot high-speed test track used for simulation of high vibration and acceleration environments. Ballistic missile inertial guidance systems and components have been tested in this environment. Although track testing is not normally required for aircraft navigation systems, it is available for special purpose systems.

1.2.5 The Directorate of Materiel provides aircraft maintenance and modifications as required to support inertial navigation system testing.

1.2.6 The Directorate of Aircraft and Missile Test operates the aircraft and provides flight crews for in-flight evaluation of navigation systems.

1.2.7 The US Army's White Sands Missile Range (WSMR) is the free world's most versatile overland range. It provides unequaled instrumentation facilities and a security not obtainable on overwater ranges. In addition to performing precision radar and optical tracking of flight tests, WSMR performs tracking data reduction and acts as lead range in scheduling tests requiring the use of the facilities of other ranges, such as Fort Huachuca, Arizona, the Western Test Range, Vandenberg AFB, California, and the radars at Edwards AFB, California.

1.3 Types of Tests

Tests conducted at the CIGTF include testing of pure (unaided) inertial, doppler inertial, stellar inertial, stellar-inertial-doppler, and doppler heading reference systems. A breakout of these tests is included under paragraph 2, Test Approach. Test programs to accommodate other systems' concepts, such as inertial-LORAN-D, will be formulated at a later date.

1.4 Future Plans

1.4.1 Any test facility must constantly plan to improve its capability in order that more advanced test specimens may be evaluated. At the CIGTF, capability improvements being considered are the acquisition of higher performance test beds, extension of flight times by use of other ranges, the use of additional tracking facilities on WSMR, and the inclusion of other methods in new programs.

1.4.2 The first flight test conducted at the CIGTF made use of a C-131. Two F-106s and a C-130 were acquired later, so that several systems could be tested simultaneously. Further improvements in test capability will result when aircraft with performance envelopes similar to the C-135 and F-4C are available as test beds.

1.4.3 Similarly, the use of other test ranges will extend the test capability of the CIGTF. In addition to WSMR, the tracking facilities at Green River, Utah and Fort Huachuca, Arizona have been used in inertial navigator flight tests. Continuous coverage can be obtained on long-range flights using the radars at Edwards AFB, California and the Western Test Range, Vandenberg AFB, California.

1.4.4 In addition to the FPS-16 radars, which provide precise continuous position tracking, such systems as DOVAP and cinetheodolite nets are available and will be used in future tests for velocity measurements and for more accurate position measurement.

1.4.5 Van testing shows significant promise in the initial testing of inertial navigators and in the evaluation of mechanization techniques. This method of testing is economical, and provides the opportunity to frequently check platform alignment optically during a test. In addition, velocity damping and accurate position updating can easily be applied to the system. WSMR, with its many precisely surveyed bench marks, is ideal for van testing, and the CIGTF is developing a van test capability.

2. TEST APPROACH

2.1 Test Concept

2.1.1 In the past, inertial navigator systems were flight tested by periodically comparing system computed positions with ground check points, or by simply recording the total accumulated error at the completion of a flight. In neither case was particular attention given to the effect of the test flight path on system errors. The concept of flight testing followed at the CIGTF differs significantly from this earlier approach.

2.1.2 Inertial navigation system errors are not linear functions of time. Instead, the system errors propagate with the characteristic Schuler frequency. The sinusoidal nature of this error invalidates the technique of interpolating between errors measured at relatively widely separated points in time. In addition, the period of the Schuler oscillation is 84 minutes. Flights of approximately this duration generally exhibit a misleading low terminal error because the Schuler component of system error is a minimum at this flight time.

2.1.3 System errors are also dependent on the parameters of the test flight path because the various error sources respond differently to various applied accelerations. It is therefore necessary to design flight paths which decorrelate error sources for individual evaluation.

2.1.4 Some consideration must also be given to the accuracy of the reference to which system performance is compared. Optical or photographic observation of ground check points is limited in accuracy, not only by the precision with which the location of the check point has been determined, but also by uncertainties and changes in camera attitude.

2.1.5 The test technique employed at the CIGTF is based on continuously measuring aircraft position during flight, using radars situated at precisely surveyed locations. Flight paths used in the tests have been computer-designed to decorrelate the various system error sources.

2.1.6 Continuous measurement of system errors also enables the evaluation of the system performance model. The performance model is a mathematical relation between the input to the system and its output, in terms of the individual error sources. Evaluation of the coefficients of the performance model therefore identifies the dominant sources of system error. In addition, determination of the complete performance model permits simulation of system operation and determination of resultant error under various operating conditions and missions. This determination of a system performance model is not a requirement under the verification program, but is added information which better allows the inertial navigator buyer to choose the proper system for his application.

2.1.7 If a system is being considered for a specific mission application, a complete evaluation requires observation of its performance under the peculiar conditions of that mission. Examples are the requirement for rapid warm-up and take-off, in-flight alignment, and high-speed flight at low altitude. If these are conditions of its ultimate mission, they should be included in the complete test program.

2.2 Test Programs

2.2.1 A well-planned program should provide for thorough testing throughout. The early stages of the program should include competitive testing of the components being considered for use in the inertial navigator. Such tests frequently reveal design inadequacies, which, if uncorrected, would result in acquisition of an unsuitable system. These tests should obviously be scheduled early enough to avoid overall program delays, should re-design be required. Similarly, evaluations of the integrated system should be conducted well before the scheduled commitment of the inertial navigator to its operational vehicle.

2.2.2 Two forms of system testing are performed at the CIGTF. Developmental testing is conducted on early prototype equipment to provide a basis for design improvement, as well as to evaluate performance. Verification tests are performed on systems which are well along in the development cycle and normally have undergone some previous dynamic testing.

2.3 Contractor Support

2.3.1 The concept of operation for aircraft inertial navigation systems tests at the CIGTF is "in-house". Air Force personnel not only manage and conduct the programs, but they also maintain (with contractor support) and operate the system. Successful operation and maintenance are dependent upon two factors:

(1) A minimum of two weeks schooling on the system is required at the contractor's plant. This training is normally contracted for and funded by Air Training Command for three officer engineers and three airmen technicians. The two groups receive separate training. The officers' course includes instruction on theory of operation, mechanization, and operating and maintaining the system, with additional emphasis on system analysis. The technicians' effort is directed toward system operation and maintenance. In the case of more complex systems, longer training periods and additional Air Force personnel will be required.

(2) Contractor technical and spares support is required at the CIGTF. Obviously, Air Force personnel cannot hope to be experts on a specific system after a minimum amount of training; thus, two or three contractor personnel are required to support the test effort. Contractor personnel do not fly on test aircraft, nor do they become actively involved in the conduct of the test program. Spares support is necessary to aid

completion of the tests in a timely manner. Systems tested at the CIGTF should have a mirror or cube mounted on the inner element of the inertial reference unit. The mirror will be used to externally monitor azimuth alignment before flight and to check azimuth after the flight.

2.3.2 Test programs for various types of systems are shown in Tables I, II, and III. The flight program is dependent upon a specific number of good data flights rather than a specific time period. The time shown in the tables is based on estimates and is a function of many variables beyond the control of the testing agency. Contractor support, therefore, should also be based on completion of the total flights and not a specific time period. In order for the CIGTF to verify a specific system, as referred to in the DOD memorandum, it will be necessary to complete the full flight test program (Phases I-A, II-A, and III-A).

2.3.3 Component testing (gyros and accelerometers) should be scheduled so that the test data are available well in advance of the system tests. In addition to providing any necessary re-design information, component parameter values and day-to-day shifts in these values are particularly useful in analyzing system performance and in predicting required system calibration intervals. (See Appendix B.)

2.3.4 System environmental tests are also outlined in Appendix B. Again, test times vary depending on customer requirements. Normally, environmental testing should follow system flight tests in order not to interfere with the flight test schedule in case of system breakdown in severe environments. If two systems are available, the environmental testing can parallel the flight tests, thereby reducing the total test time.

TABLE I

PURE (UNAIDED) INERTIAL NAVIGATION SYSTEM PROGRAM

CATEGORY	PHASE I-A STATIC TEST	PHASE II-A C-130	PHASE III-A F-106	PHASE III-A C-130
Verification	3 Weeks	12 Flights 53 Hours 1.5 Months	15 Flights 20 Hours 2.5 Months	15 Flights 45 Hours 2 Months
Developmental	2 Months	20 Flights 88 Hours 2.5 Months	30 Flights 40 Hours 4 Months	25 Flights 75 Hours 3.5 Months

Notes: (1) CIGTF preparation time prior to system delivery is normally 4 to 6 months.

(2) Helicopter test times for Phase III will be determined later.

(3) Developmental testing schedules will remain flexible to allow for the many unknowns in this type of test program.

TABLE II

DOPPLER HEADING REFERENCE, DOPPLER INERTIAL,
SINGLE AIDED INERTIAL, OR STELLAR INERTIAL NAVIGATION SYSTEM TEST PROGRAM

CATEGORY	PHASE I-A STATIC TEST	PHASE II-A C-130	PHASE III-A F-106	PHASE III-A C-130
Verification	1.2 Months	16 Flights 72 Hours 2.5 Months	25 Flights 33 Hours 4.3 Months	20 Flights 60 Hours 4 Months
Developmental	2.5 to 3 Months	25 Flights 112 Hours 4 Months	50 Flights 67 Hours 6 Months	33 Flights 100 Hours 5 Months

Notes: (1) CIGTF preparation time prior to system delivery is normally 4 to 6 months.

(2) Helicopter test times for Phase III will be determined later.

(3) Developmental testing schedules will remain flexible to allow for the many unknowns in this type of test program.

TABLE III

STELLAR-INERTIAL-DOPPLER NAVIGATION SYSTEM TEST PROGRAM

CATEGORY	PHASE I-A STATIC TEST	PHASE II-A C-130	PHASE III-A F-106	PHASE III-A C-130
Verification	1.5 Months	20 Flights 100 Hours 3 Months	35 Flights 50 Hours 5.5 Months	25 Flights 125 Hours 4 Months
Developmental	3.5 Months	25 Flights 125 Hours 6 Months	55 Flights 75 Hours 8 Months	40 Flights 160 Hours 6 Months

Notes: (1) CIGTF preparation time prior to system delivery is normally 4 to 6 months.

(2) Helicopter test times for Phase III will be determined later.

(3) Developmental testing schedules will remain flexible to allow for the many unknowns in this type of test program.

3. STANDARDIZED TEST

3.1 Test Philosophy

3.1.1 To comply with the DOD memorandum, all aircraft inertial navigation systems will be subjected to a standardized test series, Phases I-A, II-A, and III-A, plus any special tests determined by the CIGTF or the customer, Phases I-B, II-B, and III-B. The total verification program for a pure inertial system will be as follows:

Phase 0	Customer Pretest
Phase I-A	Static Integration Test (3 weeks)
Phase I-B	Special Test
Phase II-A	C-130 Airborne Standard Test (12 successful data flights - approximately 6 weeks)
Phase II-B	Special Test
Phase III-A	Aircraft Operational Performance Test (utilizing fighter, transport, or helicopter aircraft as required; 15 successful data flights - approximately 11 weeks)
Phase III-B	Special Test

Phase 0 will be accomplished by the customer to determine system suitability prior to verification tests at the CIGTF. Phases I, II, and III will be conducted at the CIGTF. Each phase will be divided into two parts. Part A will consist of standardized tests that will be performed on all aircraft inertial navigation systems for evaluation and analysis. Part B will consist of special tests that may be required to verify any particular system capabilities and requirements, or operational testing not covered by the standard test (Part A).

3.1.2 Phase 0. The customer pretest is conducted at a government or contractor facility. It is desirable that a CIGTF representative witness the final pretest.

3.1.3 Phase I-A. The system is integrated with instrumentation and all interface problems are solved. System calibrations and five static navigation runs are performed. During this phase, basic system parameters are obtained, as well as an indication of system performance.

3.1.4 Phase II-A. This phase is conducted in a transport (C-130) aircraft. The system is flown in a relatively benign environment and is completely accessible to test personnel at all times. This phase begins with two shakedown flights to prove system and instrumentation operation, and progresses to 12 fully instrumented data collection flights.

3.1.5 Phase III-A. This phase is conducted in either transport (C-130), fighter (F-106), or helicopter aircraft, depending on the intended operational application of the inertial system. Fifteen successful data flights will be conducted. These flights are fully instrumented for data collection and system evaluation.

3.1.6 Phases I-B, II-B, and III-B will be the special, negotiable tests that may be required to verify any particular system capabilities, etc., not covered by the standard test (Part A). Phase III-B will include operational tests in which flights will be made with the environment and profile as close to operational flight as possible. It is desirable that these be conducted at the CIGTF, however, in tests for other services, it may be expedient to conduct these tests at the sponsoring agency's facility to take advantage of particular operational test beds. In this event, the CIGTF test engineer will closely monitor the program and will use the results in the CIGTF final evaluation.

3.1.7 The estimated test times, shown in the phase outline (paragraph 3.1.1), are based on optimum test conditions. The number of successful data flights achieved during a test phase determines the completion of the test phase (II and III). A total of 27 successful data flights is required to complete the standardized verification program. A successful data flight will be determined by the CIGTF. It does not include any flight whose data is adversely affected by aircraft or instrumentation failures or malfunctions.

3.1.8 System verification will consist of completing Phases I-A, II-A, and III-A (the standardized test), plus other tests agreed to by the customer included under Phases I-B, II-B, and III-B. In addition, other tests could be included in Part B of any phase, which may not be required for verification.

3.1.9 Any major change in a system after verification (change of component type, computer, etc.), as well as any basic design performance change, such as accuracy or reaction time, will require re-verification. Additionally, when system development progresses to production models, a short test should be conducted to verify consistency of production.

3.1.10 After completion of the C-130 Phase III-A tests, if the CIGTF believes a system demonstrates other possible applications, F-106 and/or helicopter aircraft testing will be proposed. This will assure completeness of test results for future DOD applications.

3.1.11 After flight testing, or in parallel if two systems are available, an environmental laboratory test may be conducted to determine the tolerances of the system to vibration, acceleration and shock, and to changes in ambient temperature, humidity, and atmospheric pressure. The facilities of the CIGTF permit the evaluation of the response of a system to single or combined environmental conditions.

3.1.12 In addition to the standardized test, under Phase III-B, the CIGTF will assist in arranging customer flight testing in an operational aircraft using WSMR range and CIGTF instrumentation facilities.

3.2 Test Objectives

3.2.1 Phase I-A. The objectives of this phase are to:

(1) Verify complete system-can weight and size.

(2) Verify satisfactory system operation after delivery to the CIGTF, and measure the power required to operate the system.

(3) Integrate the system and test instrumentation.

(4) Calibrate the system (parameters) and gather system data from which to establish repeatability of gyro and accelerometer parameters.

(5) Evaluate static system navigation accuracy.

(6) Acquire system reaction time data.

(7) Acquire limited data on maintainability, reliability, and operational suitability.

3.2.2 Phase I-B. The objectives required for any special test due to system peculiarities or customer request.

3.2.3 Phase II-A (C-130 Flight Test). The objectives of this phase are to:

(1) Verify proper system integration with test instrumentation and aircraft.

(2) Obtain system performance data.

(3) Maximize the propagation of system errors for use in system analysis.

(4) Determine error coefficients for the dominant sources of system error.

(5) Acquire limited maintainability, reliability, operational suitability, and system reaction time data.

3.2.4 Phase II-B. The objectives required for any special test due to system peculiarities or customer request.

3.2.5 Phase III-A (C-130, F-106, or Helicopter Flight Test). The objectives of this phase are to:

(1) Verify proper system integration with test instrumentation and aircraft. (F-106 and/or helicopter flight tests only.)

(2) Obtain system performance data.

(3) Determine system accuracy to include error coefficient recovery.

(4) Predict operational system accuracies utilizing Phase II-A error coefficients.

(5) Complete the flights necessary for verification.

(6) Acquire limited maintainability, reliability, operational suitability, and system reaction time data.

3.2.6 Phase III-B. The objectives required for any special test due to system peculiarities or customer request.

3.3 Test Procedures

All system operation that is intended to produce primary test data in Phases I-A, II-A, and III-A will be accomplished by CIGTF personnel. Additionally, CIGTF project personnel will make the final decision on any question involving the validity or usability of all test data, as well as all day-to-day operational questions. This will include, but not be limited to, decisions on when a system will fly and when a system is out for maintenance.

3.3.1 Phase I-A

(1) Initial Checkout. A complete visual check of the system will be accomplished to insure against possible damage during shipment. The complete test specimen, including signal conditioning and the container can, will be weighed to insure compliance with the 300 pound maximum acceptable weight*.

(2) System Operation and Power Check. Upon completing the initial checkout, power will be applied to the system. A functional check of the system is performed. System outputs, such as those from the gyros, accelerometers, and computer, are checked to determine normal operation. System power requirements are measured during system operation.

(3) System and Instrumentation Integration. Flight test instrumentation will be connected and operated in conjunction with the inertial navigator to insure proper integration and operation. System output will be recorded on the magnetic tape recorder and the resulting test tape will be delivered to the Directorate of Technical Support to verify the adequacy of the data reduction plan.

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* Although 300 pounds is considered maximum for F-106 weight and balance, if greater weights are mandatory, coordination should be effected with the CIGTF as soon as possible.

(4) System Calibration. Each system will be calibrated at least three times during Phase I-A. Additional calibrations will depend on the repeatability of the data during the first three calibrations. (See Appendix E for a sample calibration procedure.) The parameters to be determined during the system calibration are gyro and accelerometer misalignment angles, accelerometer bias and scale factor, and gyro null bias and wheel-on drift rate. The system calibration will be performed on a tilt table, if required. Contractor technical representatives should be present for consultation and recommendations; however, they will not actively perform the calibration.

(5) Navigational Tests. Pre-flight evaluation of system navigational accuracy will consist of five, 8 hour navigation runs normally on a Scorsby table. These runs will be performed on separate days from a cold start in the ambient test area environment with the inertial platform located on the Scorsby table operating in roll, pitch, and yaw at any rate between 1/5 and 6 oscillations per minute with an amplitude of ±3 degrees (6 degrees total swing). System indicated position will be read from the display panel and will be manually recorded at five minute intervals. Radial plots of system indicated error (arc minutes) versus time (minutes) in the navigate mode will be made.

(6) Reaction Time Data. Reaction time (time from power on to navigate) will be measured on each of the navigation runs.

(7) Operational Data. Records of total system operating time, time between failures, time to change components, etc., will be kept.

3.3.2 Phase I-B. Any special test as required due to system peculiarities or customer request.

3.3.3 Phase II-A

Phase II-A will be conducted in a C-130 aircraft. The system is flown in a relatively benign environment and is completely accessible to test personnel at all times. Sufficient, qualified CIGTF test personnel will accompany the system to operate it, observe its in-flight performance, to prepare in-flight quick-look error plots, and to operate the necessary test equipment.

(1) System and Aircraft Integration Check. The system will be installed in the C-130 and all connectors for cables and cooling (if required) will be mated. External power and cooling, as required, will be applied to the aircraft. The system will be operated through its various modes and navigated for one hour to assure proper operation. During the one hour navigation run, a system output tape recording and visicorder tape will be cut and all recorded signals will be observed. A CEC recording is made from the magnetic tape for a detailed study of the signals. A 15 minute taxi check will be made after accomplishing a complete pre-flight check (see Appendix F). Both a magnetic tape and a visicorder tape will be cut during the taxi check. These tapes will be analyzed to assure proper operation of the system and instrumentation integration.

(2) System Data Flights. This phase begins with two shakedown flights (which may be counted as performance flights depending on the data and system performance) to prove system and instrumentation operation. Twelve successful data flights are required to complete this phase. (Appendix C diagrams the flight patterns to be used.)

System warm-up and pre-flight alignment will be performed in the aircraft while outdoors under existing environmental conditions, after at least a four hour power off state. (Pre-flight and post-flight procedures are outlined in Appendix F.)

(3) System Performance Evaluation. During all test flights, the system output is recorded on magnetic tape. The aircraft is tracked by range FPS-16 radar, whose output is also recorded on magnetic tape with the same time base. These two tapes are reduced and processed through the computer evaluation programs using the CDC-3600 computer facility to obtain system error plots and coefficients. (See Appendix A for detailed Analysis Methods and Evaluation Procedures.)

(4) Operational Data. All reaction times, as well as ambient temperatures, will be recorded on all flights. A complete history of system failures, times to failures, time to repair, etc., will be kept.

3.3.4 Phase II-B. Any special test as required due to system peculiarities or customer request.

3.3.5 Phase III-A

Phase III-A is conducted in either a C-130 cargo, F-106 fighter, or helicopter aircraft depending on the intended system application. These flights are fully instrumented for data collection. Each system will be calibrated at the beginning and at the end of Phase III-A. This calibration will provide an indication of possible parameter shifts.

(1) System and Aircraft Integration Check (F-106 and Helicopter only). The system will be installed in the aircraft and all connectors for cables and cooling (if required) will be mated. External power and cooling, as required, will be applied to the aircraft. The system will be operated through its various modes and navigated for one hour to assure proper operation. During the one hour navigation run, a system output tape recording and visicorder tape will be cut and all recorded signals will be observed. A CEC recording is made from the magnetic tape for a detailed study of the signals. A 15 minute taxi check will be made after accomplishing a complete pre-flight check (see Appendix F). Both a magnetic tape and a visicorder tape will be cut during the taxi check. These tapes will be analyzed to assure proper operation of the system and instrumentation integration.

(2) System Data Flights. For F-106 and helicopter test programs, this phase begins with two shakedown flights (which may be counted as performance flights depending on the data and system performance) to prove system and instrumentation operation. Fifteen successful data flights are required to complete this phase. (Appendix C diagrams the flight patterns to be used.)

System warm-up and pre-flight alignment will be performed in the aircraft while outdoors under existing environmental conditions, after at least a four hour power off state. (Pre-flight and post-flight procedures are outlined in Appendix F.)

(3) System Performance Evaluation. During all test flights, the system output is recorded on magnetic tape. An airborne timing generator will be used as the primary timing reference. On the F-106, a 12.5 kc precision oscillator is used as a backup airborne time base in conjunction with an IRIG signal dubbed on the aircraft tape recorder during the pre and post-flight phases. The aircraft is tracked by range FPS-16 radars whose output is also recorded on magnetic tape with the same time base. These two tapes are reduced and processed through the computer evaluation programs using the CDC-3600 computer facility to obtain system error plots. (See Appendix A for detailed Analysis Methods and Evaluation Procedures.)

(4) Operational Data. All reaction times, as well as ambient temperatures, will be recorded on all flights. A complete history of system failures, times to failures, time to repair, etc., will be kept.

3.3.6 Phase III-B. Any special test as required due to system peculiarities or customer request. This phase should include, but not be limited to, operational testing, such as rapid reaction and severe environmental profiles. These tests are strongly recommended, and, in the case of Air Force sponsored systems, will be conducted at the CIGTF.

3.3.7 Verification. At the completion of the test program, a total system error profile of the 25th, 50th, 75th, and 90th percentile error levels is presented.

3.4 Test Results

3.4.1 The following results from the CIGTF verification tests will be presented to the customer:

(1) Quick-look plots of system performance (available immediately after each flight).

(2) Reduced error plots of system demonstrated performance (available two weeks after each flight).

(3) A final report, verifying system demonstrated performance and containing data relative to operational suitability, maintainability, and reliability, is to be published two months after the last test flight.

(4) A supplementary report of detailed system error analysis to be published as soon as possible after the final report, for those systems designated by the CIGTF or the customer, which warrant this examination.

3.4.2 Quick-Look Plots. These gross indications of the system performance (position data) are obtained by time correlating the radar plotting board position information and the cockpit displayed position while in flight. Plots of this type are useful for system troubleshooting since they are available immediately after each flight. An example of a quick-look error plot is shown in Figure 2.

3.4.3 Reduced Error Plots. Complete system error performance plots will be available approximately two weeks after each test flight. See Figures 3 and 4. These plots are based on machine reduction and comparison of on-board system tapes and FPS-16 tracking tapes. A plot of percentiles of radial error for each test phase or section of a phase will be available approximately two weeks after the end of the phase or section. Velocity error information will be provided in reduced error plot form, when required.

3.4.4 Final Report

(1) A final report on system demonstrated performance will be published two months after the last test flight. This report will include, but not be limited to, an abstract, test objectives, test approach, test results, verification of demonstrated system performance, conclusions and recommendations.

(2) The test results portion of this final report will deal with the demonstrated performance and present data on operational suitability, maintenance and repair, and reliability associated with the tested system. System performance will be further defined in the following manner:

The fundamental accuracy results of the CIGTF standardized tests are in the form of a plot depicting various percentile levels of performance (see Figure 5). This plot will be general in applicability. For example, suppose that a weapon systems manager has a critical mission requirement for at most a 1.5 NM radial error throughout a mission which could last as long as three hours. He may enter Figure 5 at the 1.5 NM radial error value on the ordinate, read across horizontally until the mission time (180 minutes) is reached, and then directly interpolate the probability percentile which the system will provide for his mission. The result may be interpreted as the probability that the tested system will perform at or below the chosen radial error at that flight time. In the example, the systems manager would conclude that this particular inertial navigator could be expected to meet his requirements only 63% of the time for <u>the tested application</u>. Accuracy results from the standardized test will allow a performance comparison of verified systems flown under similar flight conditions.

(3) Additional information which will be supplied concerning system accuracy includes:

A plot of mean radial error, median radial error, and 50th percentile of radial error overplotted. These plots are included to show that all flights included in the calculations are representative of the population. Note: Flights that are not representative will be reflown.

Three composite plots of latitude error, longitude error, and radial error for each flight on a system in each test phase. These composite plots are included to show the system error patterns and repeatability of data from flight to flight.

The same error plots in the velocity domain will also be provided in this report.

3.4.5 Supplementary Report. A supplementary report will be issued as soon as possible after the final report for those systems whose test results demonstrate they warrant further detailed study. Error coefficients describing the dominant system error sources will be the primary content of this supplementary report. The error coefficients will be presented in tabular form for each flight, and the assumptions, definitions, and conclusions will be presented for the entire test program. Any areas of re-design indicated by the coefficients will be pointed out in this section, a conclusion as to system growth potential or lack thereof will be presented, and the results will be extrapolated to a long-range, long-time, high-velocity flight profile to provide an indication of system performance under those conditions. Where possible, results of an actual system flight will be used to verify the simulated results.

FIG 2

QUICK-LOOK ERROR PLOT

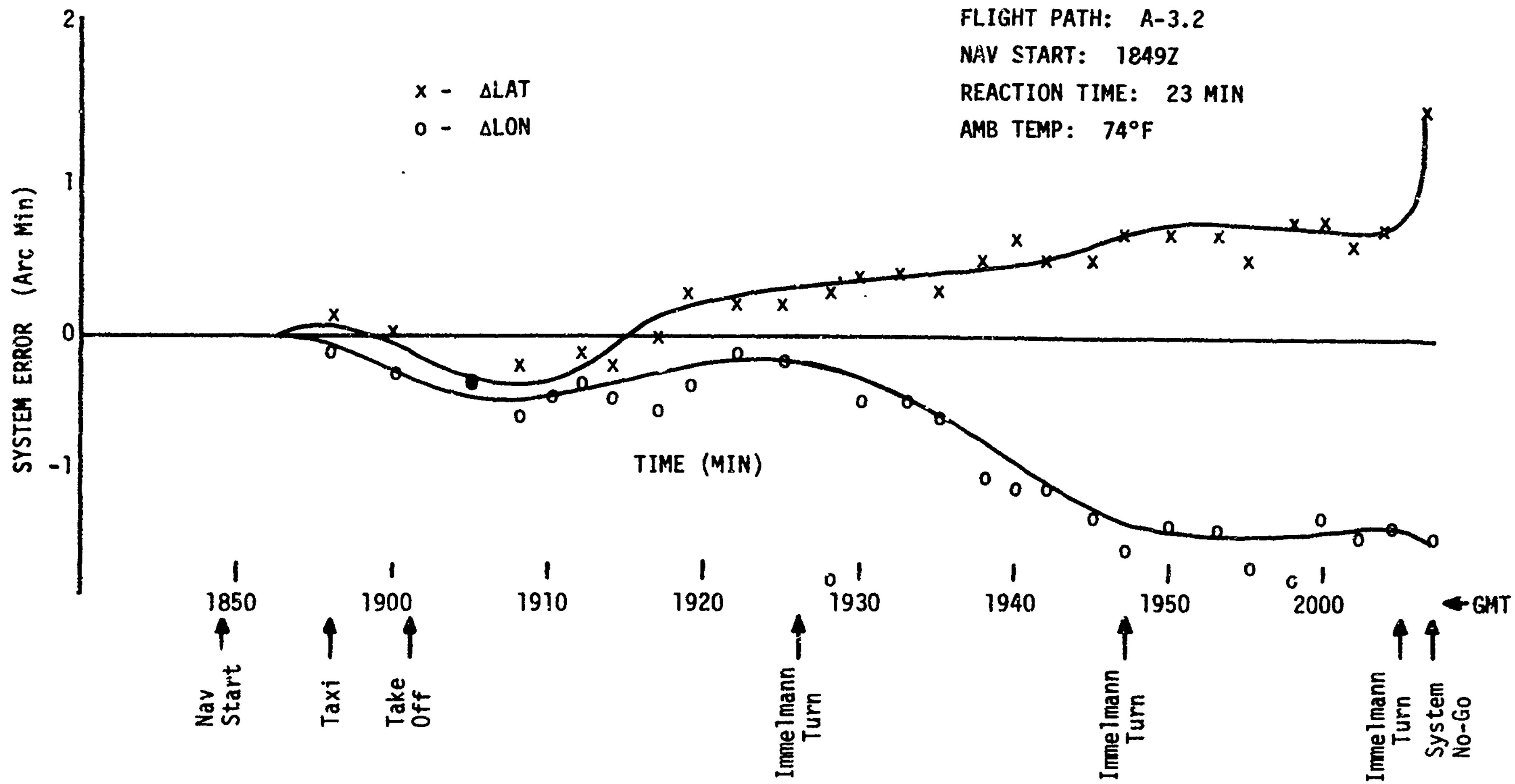


FIG 3

RADIAL ERROR

vs

TIME IN FREE INERTIAL MODE

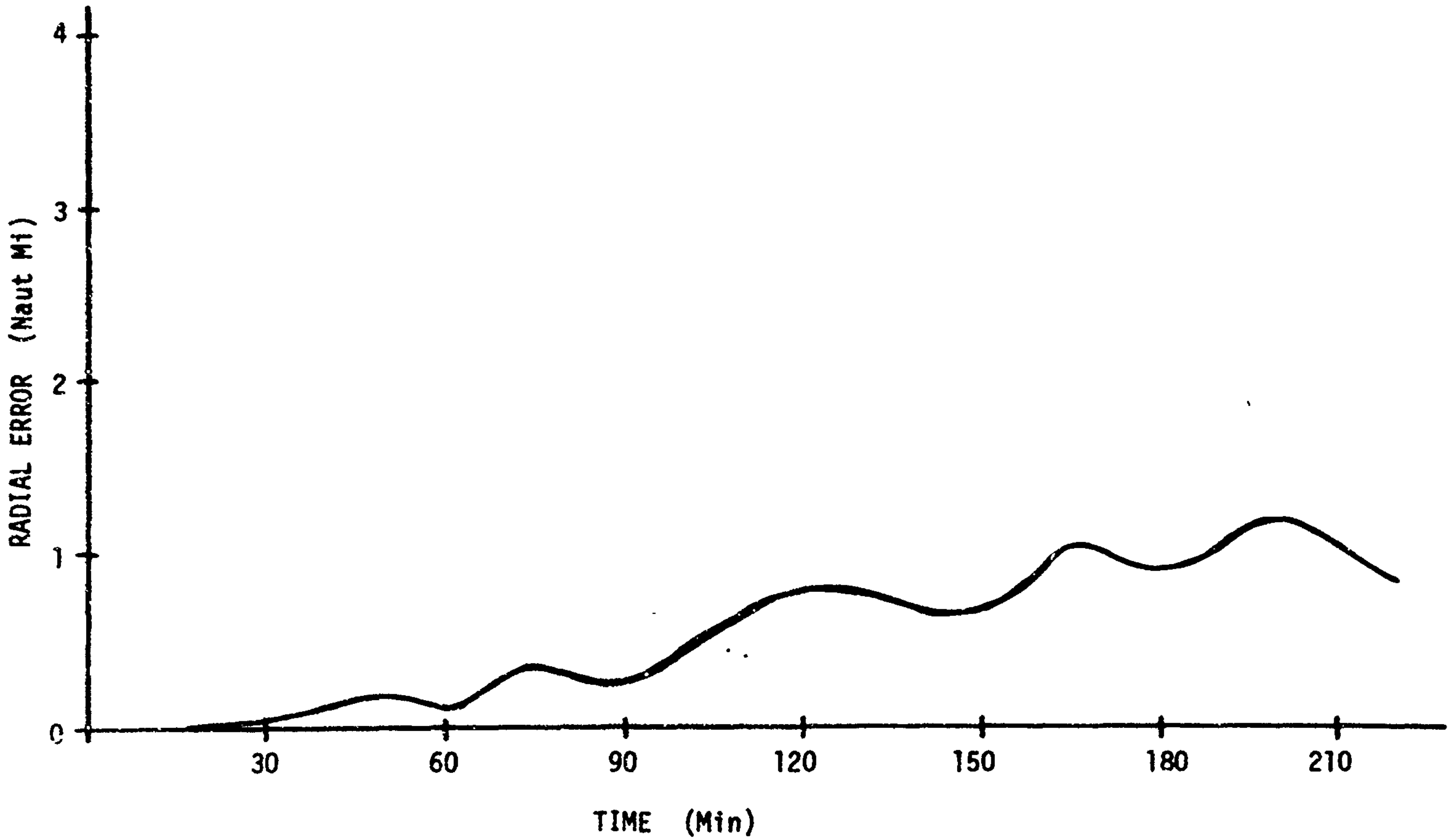
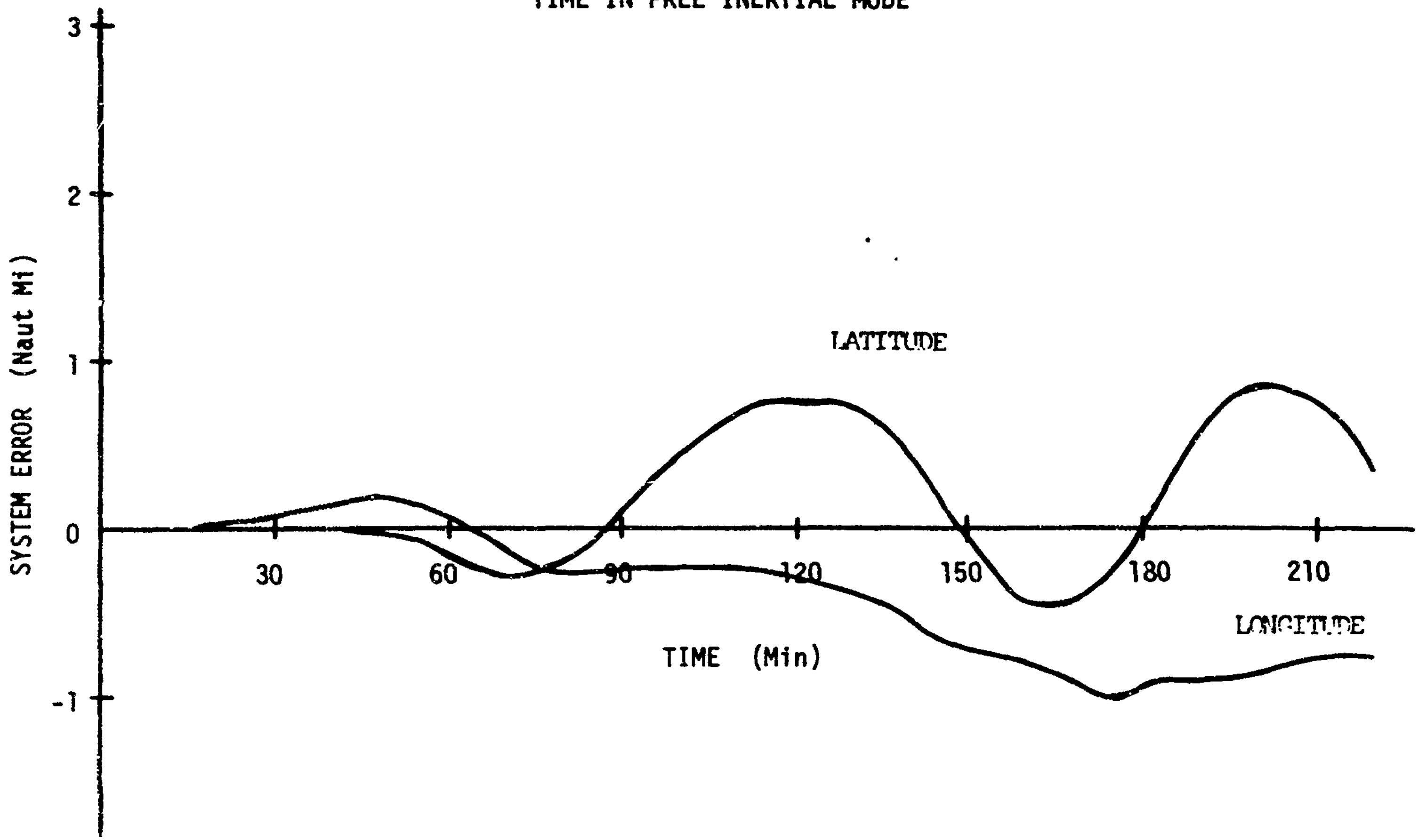


FIG 4

LATITUDE AND LONGITUDE ERROR
vs
TIME IN FREE INERTIAL MODE



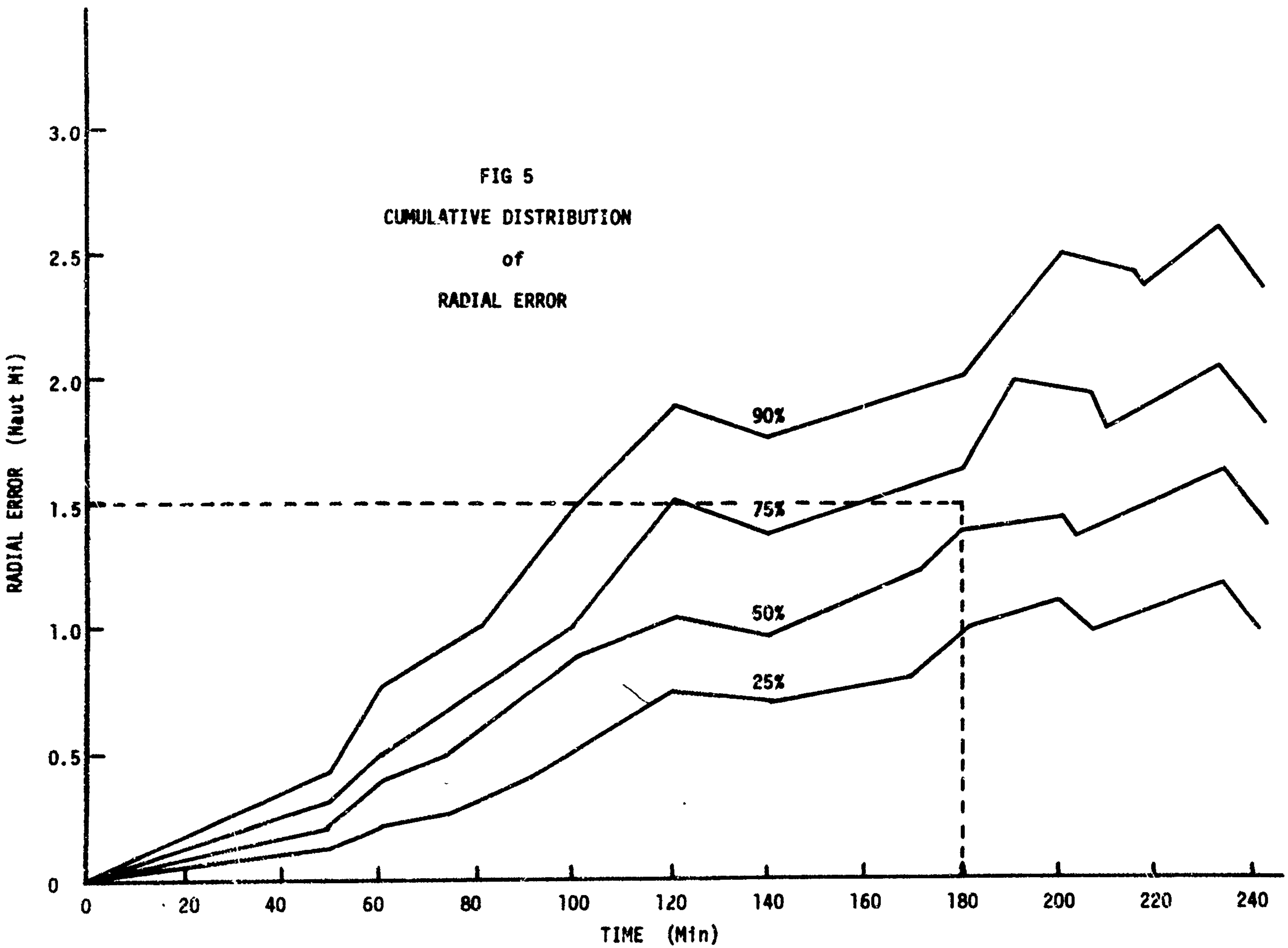


FIG 5
CUMULATIVE DISTRIBUTION
of
RADIAL ERROR

4. DATA

4.1 Collection. The following categories of data are collected during any flight test progr . (Phases I-A, II-A, and III-A):

System Size and Weight
Power Required
Component Parameters
Computer Clock Output
System Reaction Time
System Wander Angle
System Azimuth Alignment (if possible)
Failures and Malfunctions
Maintenance Required (including parts' replacements)
Maintenance Time
Individual Part's Replacement Time
System Operating Time
System Position (Indicated)
Aircraft Position (True)
System Velocity (Indicated)
Aircraft Velocity (True)
Power Supply, Voltage and Frequency
Altitude
Environmental Temperature and Pressure
Environmental Three-Axis Vibration Levels

4.2 Processing

4.2.1 All data are processed and controlled by Air Force personnel (except range tracking data presently processed by the US Army at WSMR).

4.2.2 Quick-look error plots are produced during the mission by comparing system indicated position with the aircraft position shown on the radar plotting board. These plots permit qualitative evaluations of performance during each flight and indicate system malfunctions requiring correction before subsequent tests.

4.2.3 Following the mission, formal reductions of the data recorded on board the aircraft and at the ground station are performed. Using radar data as reference, system error as a continuous function of time is derived. Statistical analysis tec niques are then applied to this function to determine the performance model coefficients.

4.3 Distribution

4.3.1 Initial distribution of all data and test results will be controlled by the CIGTF. Distribution lists (designated by the customer/CIGTF) will be contained in the specific system test plan. Customer proprietary rights will be observed.

4.3.2 Test data and preliminary test results will be available to the customer as soon as possible after each test. Normally, a quick-look error plot, to include any significant occurrences, is available immediately after each test event.

4.3.3 A final report is furnished to the customer within about two months following completion of the test program.

4.4 Classification. Test data and results will be accorded a security classification commensurate with the program and the system under test.

5. PERFORMANCE ANALYSIS

Inertial navigation systems submitted to the CIGTF for evaluation are given the closest scrutiny and most detailed error analysis possible consistent with the limitation of the test program length. Systems are analyzed for performance accuracy and for merit in the areas of reliability, maintainability, and operational suitability. Appendix A, Analysis Methods and Evaluation, describes the methods and analytical procedures for a fine-grain error analysis of systems tested by the CIGTF.

6. SUPPORT EQUIPMENT

6.1 Ground Equipment

6.1.1 Contractor AGE peculiar to the system will be provided and be maintained by the contractor.

6.1.2 Range tracking aids will be provided as required by the WSMR. Continuous digital tape information is normally required for all flights.

6.1.3 An IRIG-B reference timing source is provided by WSMR for all missions.

6.2 Airborne Equipment

6.2.1 C-band transponders are installed in the test bed aircraft and are used as radar tracking aids.

6.2.2 An on-board timing generator, capable of operating with an error of less than one millisecond per 24 hours, provides the airborne timing reference signal. A secondary timing reference for fighter flights is provided by a precision oscillator contained in the magnetic tape recorder electronics. This signal, in turn, is referenced to IRIG-B time by recording three to five minutes of IRIG-B time from a ground timing source on the magnetic tape before and after the flight.

6.2.3 Data is recorded on airborne magnetic tape recorders and airborne oscillograph recorders. (See Appendix D for details.)

(1) The tape recorder is used primarily to record system position and velocity data, a timing reference signal, and three-axis vibration levels for subsequent data reduction and analysis.

(2) The oscillograph recorder is used to record quick-look system operation data and system troubleshooting signals. Aircraft power and cooling parameters are also recorded.

6.2.4 Display panel cameras are used to record quick-look system information if the amount of data required or type of mission flown prevents the system operator from manually recording system data.

7. SYSTEM INSTALLATION

7.1 To meet aircraft compatibility and interchangeability requirements, the CIGTF uses contractor supplied system container cans for flight testing inertial navigation systems. This approach gives the flexibility between transport and fighter aircraft required to obtain maximum flight test data in a minimum time period.

7.2 During transport testing, the system containers are mounted in racks built by the CIGTF to standardize (as much as possible) mechanical and electrical interfaces with the aircraft. As the system progresses to fighter test bed aircraft, these same system cans are installed in either the left or right forward electronics bay area of an F-106A or B. A universal can has been designed such that a system can be installed as either a left or right hand system. System cans are interchangeable between F-106A and B model aircraft provided the air cooling duct location is shifted a few inches.

7.3 Systems with special mounting space or electrical requirements, which prevent the contractor from conforming to the standard can configuration, should be discussed on an individual basis with the CIGTF. Specific wiring details should be coordinated with the CIGTF to insure compatibility.

7.4 The can concept discussed here has been developed for flight testing in currently available C-130 and F-106 aircraft. Details regarding helicopter system installation will be formulated when a particular helicopter is obtained. Additionally, in the future, when new test bed aircraft are obtained, exact design specifications may vary.

7.5 Power. The following maximum power is available for each system can.

dc - 28 volts, 280 watts
ac - 115/208 volts, 3 phase, 400 cycle, 2.5 KVA
Power switch over time is 50 ms or less.
Mil-Std 704 applies

7.6 Cooling

7.6.1 F-106 Aircraft

(1) Flow Rates. During ground operation, the minimum air flow rate to each system in the electronics bay is approximately 18-1/2 pounds per minute. During flight, this flow rate will decrease with altitude, reaching a value of about 10 pounds per minute at 40,000 feet.

(2) Temperature. During both ground and flight operation, air is delivered to the electronics bay at 55° ± 5°F.

(3) Pressure. At engine idle the minimum cooling air pressure is 2.5 inches of water.

7.6.2 C-130 Aircraft. Cooled air is not provided on a regular basis for systems flying in the C-130, but reasonable power for auxiliary air blowers can normally be made available. The cabin temperature in the C-130 can reach 115°F on a hot summer day.

7.7 Can Construction

7.7.1 Weight. The total weight of the can and system should not exceed 300 pounds.

7.7.2 Size. Can size should conform to the dimensions of Figure 6.

7.7.3 Structure. Even though an open-sided box is indicated in Figure 6, a frame or truss configuration may be substituted. All cans will be hardmounted to the aircraft structure and will not rely on the aircraft for structural strength or rigidity. The internal configuration of the can will be determined by the contractor. See notes on Figure 6 for additional information.

8. RESPONSIBILITIES

8.1 CIGTF Responsibilities. The CIGTF will:

8.1.1 Provide program management to include documentation of the program, development of the test plan, customer coordination, conduct of all phases of the test program, and timely availability of all test data and results.

8.1.2 Provide test support to include the physical plant, aircraft, range, instrumentation, and data reduction.

8.2 Customer Responsibilities. The customer will:

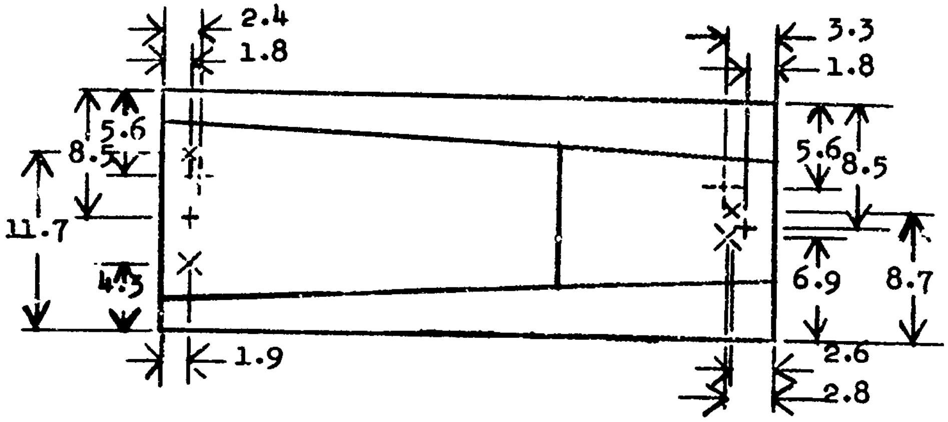
8.2.1 Provide test specimen(s), properly contained (see paragraph 7, System Installation) and properly conditioned (see Appendix D, Instrumentation) at a time agreed upon by the CIGTF and the customer.

8.2.2 Provide necessary spares and contractor technical service personnel.

8.2.3 Provide for contractor training of CIGTF personnel at the contractor's facility. This training is normally scheduled and funded for by Air Training Command (ATC).

8.2.4 Provide components for component test if required and agreed upon by the CIGTF and the customer.

All Dimensions are in Inches



NOTES: 1. Typical plug connector. Usable space for plugs is approx 12" x 8". Horizontal plate as shown.
2. Typical air duct attachment (shown for right hand F-106A position). Flexible duct allows bracket attachment to either side of F-106A, B. Brackets (2 needed, one left and one right) can be moved up and down on angles to which they are mounted.
3. Two sets of mounting points (can to AC) are needed as shown in top view. X = left side. + = right side.

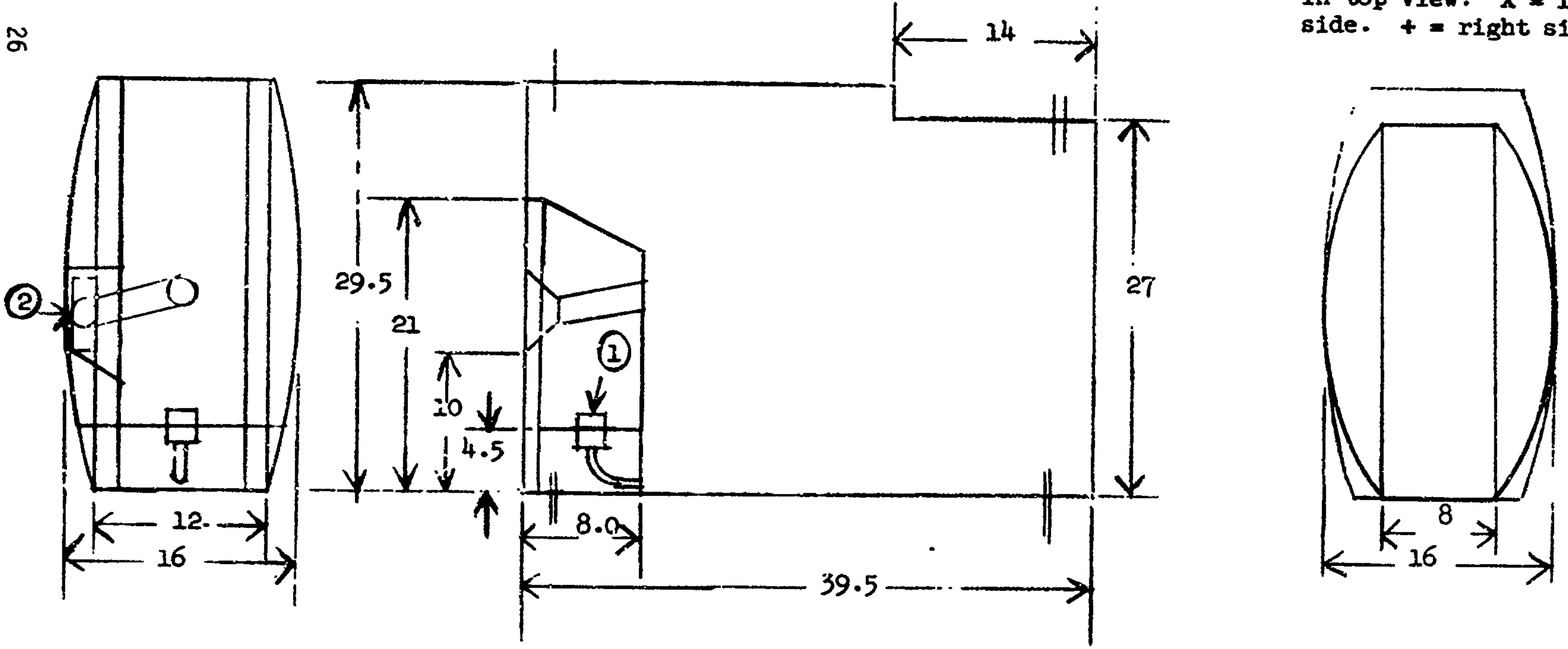


FIG 6

SYSTEM CAN

9. SCHEDULE

9.1 Lead-Time Factors (preparation time before delivery). The CIGTF requires approximately four to six months preparation time prior to delivery of a system to accomplish the following:

9.1.1 Modify the aircraft.

9.1.2 Develop data reduction programs.

9.1.3 Procure special test support equipment.

9.1.4 Document the program with AF Systems Command and White Sands Missile Range.

9.1.5 Train project engineers and technicians at the contractor's plant on system technical and operating details.

9.1.6 Program aircraft and flying hour requirements.

9.1.7 Develop the specific system test plan.

9.2 Component Testing. One to three months is required by the CIGTF for laboratory testing of gyros and accelerometers. Ideally, this testing should take place prior to final system configuration and delivery to the CIGTF for system test.

9.3 System Flight Testing. A standardized test schedule is shown in Table IV. The times describe the general chronological sequence to be followed in the verification test program. It is anticipated that two standardized test programs will be scheduled during one year. Thus, a system that misses one entry date will have a maximum of only six months to wait before entering the next scheduled test series. Normally, an additional two months after completion of the last test flight will be required for preparation of the final report. This time is not included in the standardized test schedule (Table IV). Standardized tests may be started at other times, but schedules will be coordinated with the CIGTF.

9.4 System Environmental Testing. This phase of the test program will require approximately three months. Ideally, assuming two systems are available, this could be accomplished concurrently with flight testing. Otherwise, it will take place subsequent to the flight test.

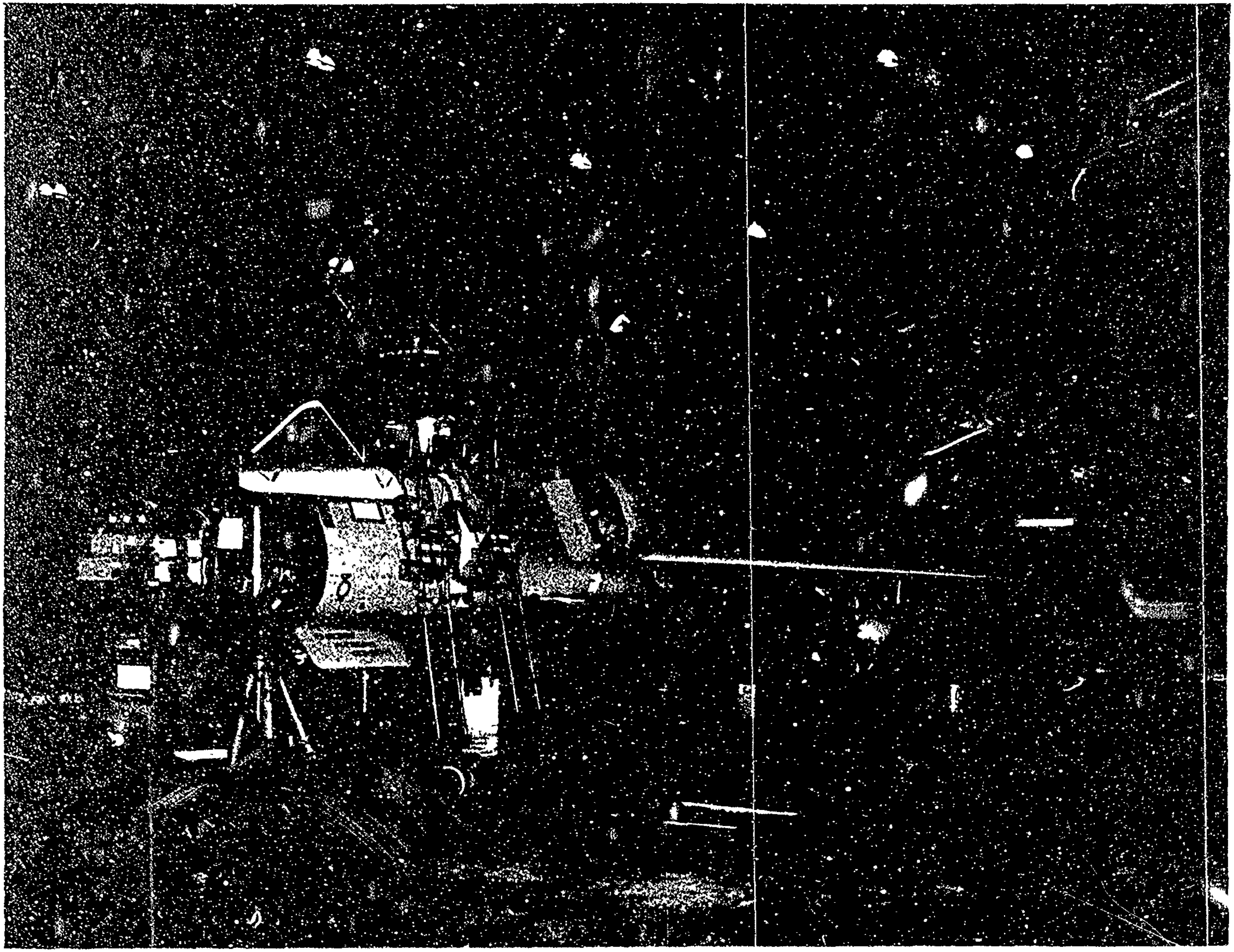
JAN	FEB	MAR	APR	MAY	JUN	JUL	AUG	SEP	OCT	NOV	DEC
	I-A	II-A	III-A Fighter*								
	I-A	II-A	III-A Cargo								
							I-A	II-A	III-A Fighter*		
							I-A	II-A	III-A Cargo		

STANDARDIZED TEST SCHEDULE
(Unaided Inertial Systems)

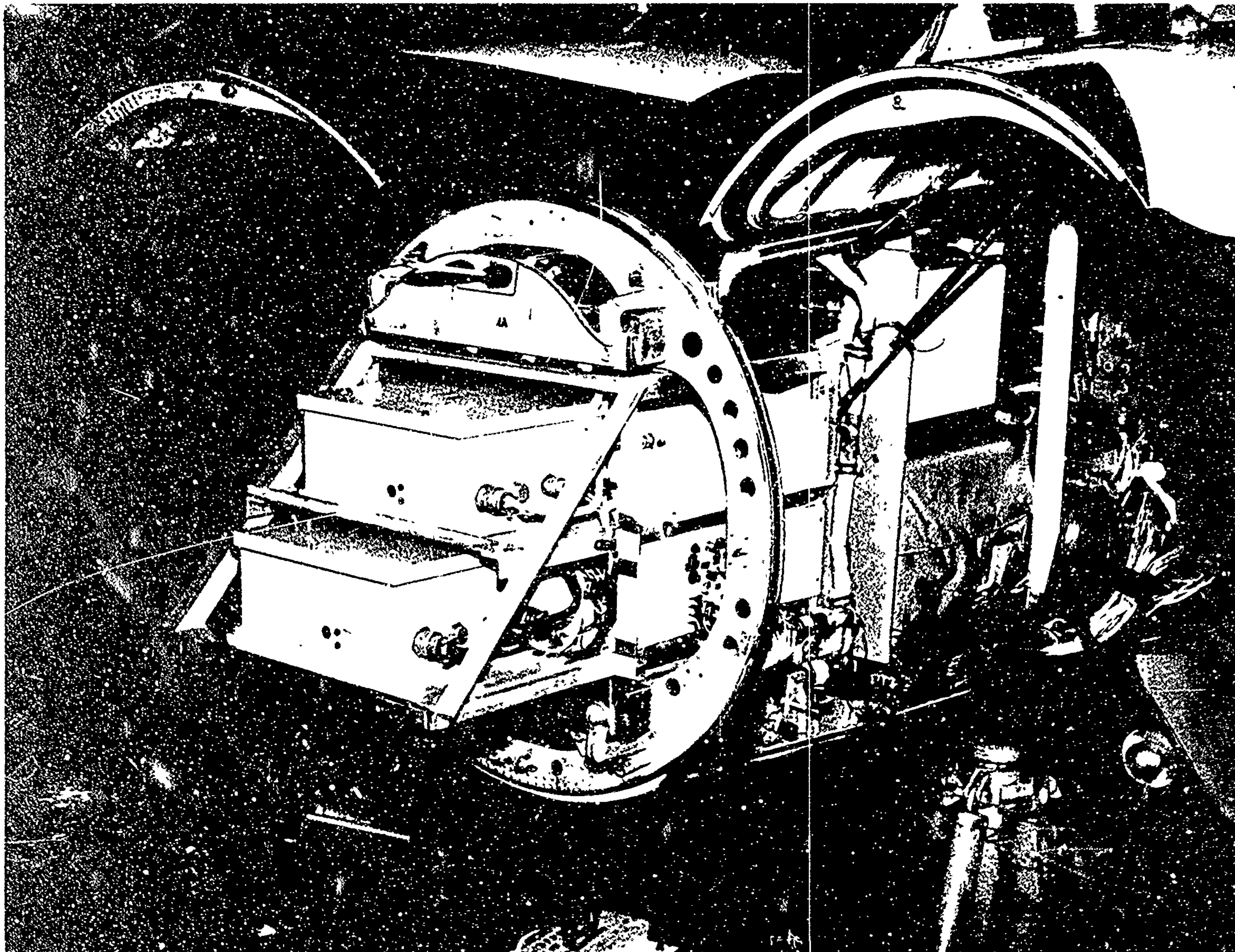
Table IV

*Same approximate time for helicopter.

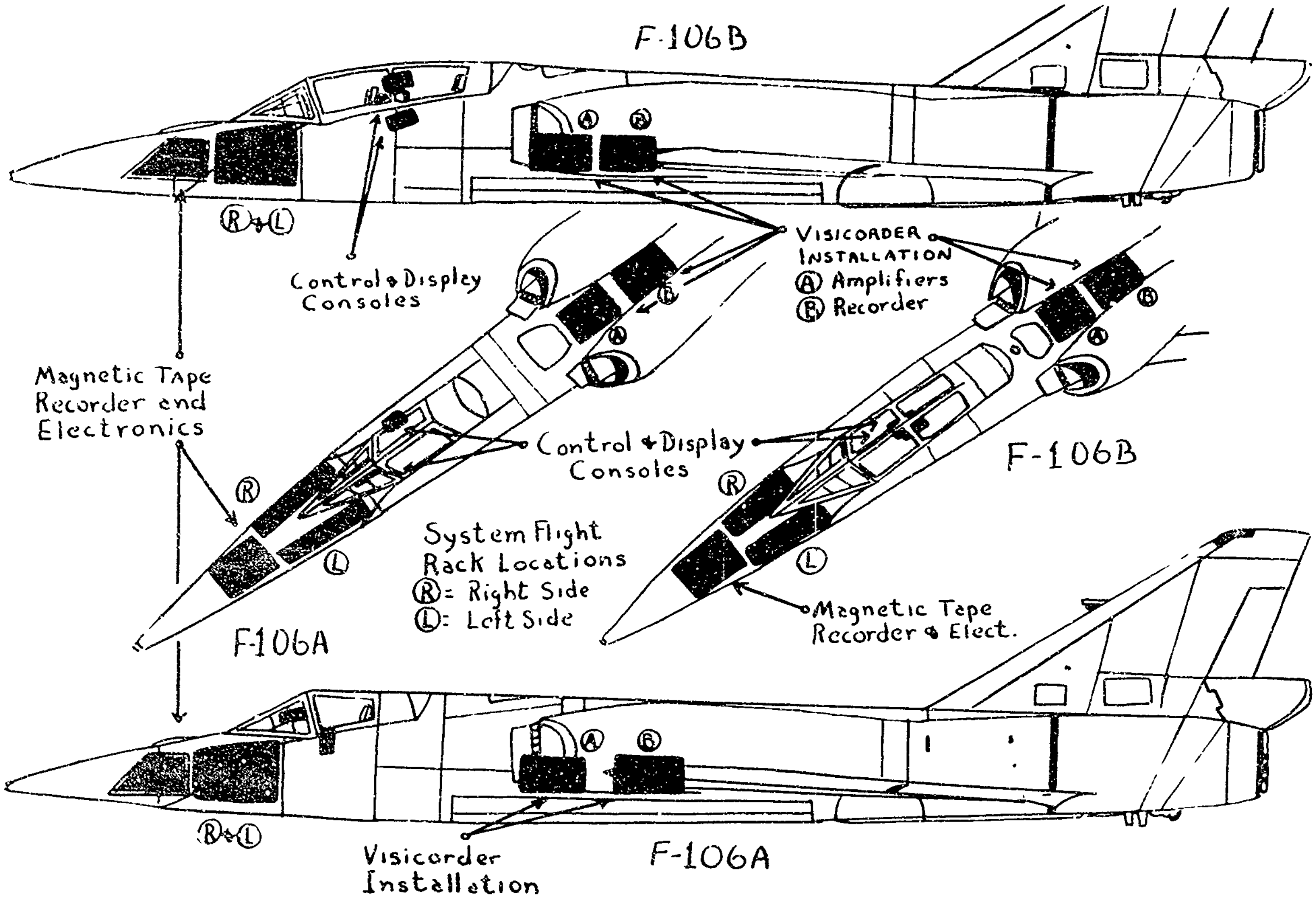
Phase II Transport Aircraft, C-130A



Phase III Fighter Aircraft, F-106B



System and Recorder Installation, F-106



Test Equipment Locations, F-106B

APPENDIX A

ANALYSIS METHODS AND EVALUATION

1. INTRODUCTION

1.1 As previously described in the Test Concept, the general test method is to fly the inertial system within the coverage of a WSMR reference (position, velocity) instrument. A time correlated error plot is made depicting the deviation of the system indicated data from the reference standard. This error plot is the basis of all system analytical studies:

1.1.1 It is used to precisely describe the system performance and relate it to specifications. (Final Report results.)

1.1.2 It is used to establish system performance parameters including error coefficients. (Supplementary Report results.)

1.2 The system accuracy evaluation provides the using agency (Air Force, Army, Navy, NASA, etc.) with much of the information required to determine whether a particular system can perform its intended mission. The error coefficients derived from the system performance can provide the user with:

1.2.1 An identification of the dominant sources of error within the system. This pinpoints areas for design improvement and provides an indication of system growth potential.

1.2.2 A measure of system performance on other flight paths and other applications. Simulations can be made which will project system performance to any flight path and any environment. This in turn allows the design of tactical profiles which will produce the minimum system error over the target.

1.3 A detailed list of the specific test results to be presented in the Final Report and the Supplementary Report is presented in Section 2 of this appendix. The data reduction procedures required to determine system accuracy for the Final Report are described in Sections 3 and 4. The additional procedures required to determine error coefficients for the Supplementary Report are outlined in Section 5.

2. TEST RESULTS

2.1 System Accuracy Evaluation. The fundamental accuracy results of the CIGTF standardized tests will be in the form of a plot for each phase depicting the 25th, 50th, 75th, and 90th percentile levels of performance. The following results will also be included in the final test report:

2.1.1 Plots of system latitude error, longitude error, and radial error versus time for each flight and each mode of system operation.

2.1.2 Plots of system latitude velocity error, longitude velocity error, and velocity error vector magnitude versus time for each flight and each mode of system operation.

2.1.3 Cumulative plots for the ensemble of test flights:

(1) Latitude error versus time for each mode of system operation.

(2) Longitude error versus time for each mode of system operation.

(3) Radial error versus time for each mode of system operation.

(4) Mean, median, and CEP position error versus time for each mode of operation.

(5) 25th, 50th, 75th, and 90th percentile velocity accuracy curves versus time for each mode of operation.

(6) Mean, median, and 50th percentile velocity error versus time for each mode of operation.

(7) Time versus position and velocity error for special flights with operational profile missions.

(8) Special plots as required (heading time histories, verticality time histories, etc.).

2.2 Error Coefficient Results. Error coefficients describing the dominant system error sources will be the primary analytical result of the test program. The error coefficients will be presented in tabular form for each flight, and the assumptions, definitions, and conclusions will be presented for the entire test program. Any areas of re-design indicated by the coefficients will be pointed out in this section, a conclusion as to system growth potential or lack thereof will be presented, and the results will be extrapolated to a long-range, long-time, high-velocity flight profile to provide an indication of system performance under those conditions. Where possible, the results of an actual system flight under those conditions will be used to verify the simulated results.

3. <u>DATA REDUCTION PROCEDURES</u> (System Performance Results)

3.1 Machines. All data reduction of the FPS-16 radar data and of the system data is accomplished on a CDC-3600 computer operated by the Directorate of Technical Support. All data reduction of cinetheodolite and DOVAP data is accomplished on the IBM 7094 computer of the White Sands Missile Range.

3.2 Data Reduction Time. The normal time from test flight to reduction of radar data is two work weeks; cinetheodolite and DOVAP require approximately three work weeks for data reduction.

3.3 Reference Accuracies. The accuracy of the range instrumentation is a strong function of target flight path, range, altitude, and airspeed. For this reasons, no one figure of merit is available for all test conditions. Table A-I lists the best accuracies attainable by the range instrumentation assuming the best possible tracking conditions.

TABLE A-I

WHITE SANDS MISSILE RANGE TRACKING ACCURACIES

Instrument	Latitude Longitude Position (Ft)	Altitude (Ft)	Velocity (Ft/Sec)	Aircraft Heading	Coverage
FPS-16 Radar	50	50-100	1	1 degree	Range Wide
Cinetheodolite	1-5	25-50	0.5	8 minutes	Limited Locations
DOVAP	0.5	10	0.1	--	Limited Locations

3.4 Reference Data Reduction

3.4.1 The FPS-16 radar data serves two purposes:

(1) It provides the input to a plotting board which accurately positions the aircraft during each test flight, assuring that the designed flight path is correctly followed. This real-time information also provides for a quick-look measure of system error.

(2) The taped digital radar data is processed to provide the reference position standard from which the final error plots are made.

3.4.2 The digital radar data is recorded at a rate of one sample per second. The reference information is obtained by smoothing the data and computing the results at ten second intervals. Recording is initiated just before the system is switched to the navigate mode and is terminated after flight when the aircraft returns to its initial position on the ramp.

3.4.3 Figure A-1 depicts the radar data reduction flow chart. Originally, the selection of the digital radar tape width and format was influenced by the availability of equipment which was compatible with a Univac computer. For this reason it is now necessary to process all radar tapes through tape conversion equipment (A) for conversion to IBM format. Range, azimuth, and elevation data are processed through the Single-Station

Radar position program (B) which calculates X, Y, Z position referenced to the CIGTF tangent plane coordinate system. This program accepts pre-flight calibration data and meteorological measurements to apply the following corrections to the raw radar data:

(1) Data shaft eccentricity errors.

(2) Non-perpendicularity of azimuth axis to elevation axis.

(3) Antenna deflection or sag.

(4) Non-perpendicularity of antenna beam to elevation axis.

(5) Radar mislevel error.

(6) Beacon delay.

(7) Refraction correction.

It also includes automatic survey, time, and data edit routines.

3.4.4 The Radar Geodetics program (C) uses X, Y, Z and time to calculate the geodetic latitude, longitude, and altitude of the system at each time point. The Merge program (D) combines multiple files of radar geodetic information into one file, eliminating time overlaps and interpolating over gaps in time and data. The Smooth program (E) takes the radar data at one second intervals and fits a quadratic or cubic to the data in a least squares sense, computing the best estimate of position at ten second intervals.

3.4.5 Velocity information is processed in a similar manner, with the exception that after CIGTF tangent plane velocities are computed in the Single-Station Solution program, they are converted to the aircraft coordinate frame in a Velocity Coordinate Transformation program.

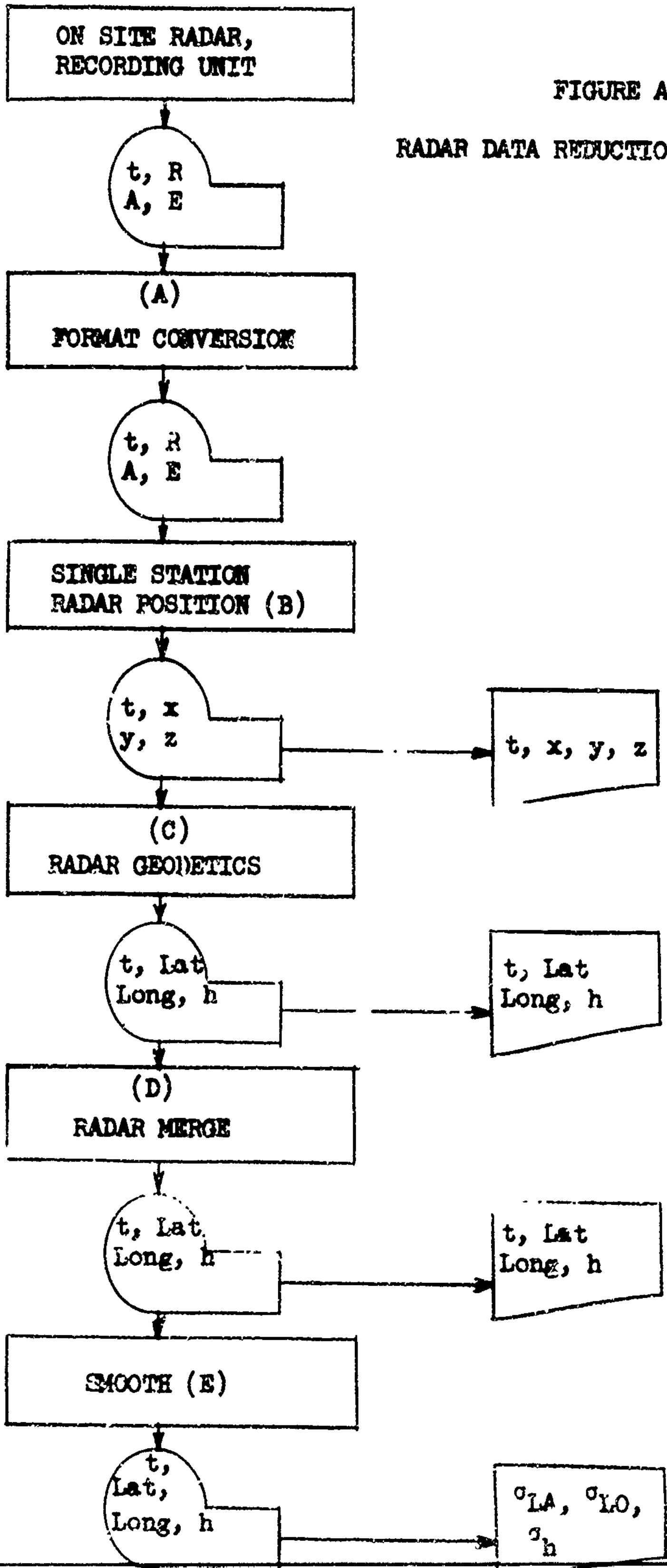


FIGURE A-1

RADAR DATA REDUCTION FLOW CHART

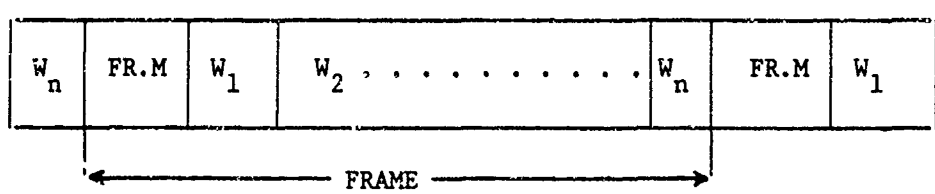
3.5 System Data Reduction

3.5.1 In order that a large number of digital systems may be tested at the CIGTF over as short a period as possible, the standard data format used by the CIGTF is included in this document. *

3.5.2 System Data Flow.** The flow chart for the system data reduction is depicted in Figure A-2. The General Input Converter (GIC) quantizes the system data tape and converts the data to a standard IBM format. The System Interpolation program scales the data, scales and biases both the reference and system time series, and linearly interpolates the data to two specified time series. The results of the System Interpolation program are output in a format compatible with the results of the radar reduction flow chart for ease of forming error vector information.

3.5.3 CIGTF Format. PCM System, i.e., a continuous flow of binary data which is separated by frame markers with each frame further subdivided into words.

Example:



(1) The frame marker and each word in the frame have the same length, i.e., the same number of bits.

The frame marker consists of a fixed pattern for the whole word and is the first word in the frame.

Suggested Pattern: 10 10 10 10 10 or

11 00 11 11 0 11 11 0 1 0 11 0 11

---

* It is realized that certain system idiosyncracies may require deviation from this format.

** For the purpose of this document, all systems are considered to have digital computers. A different method of data reduction is employed for analog mechanizations.

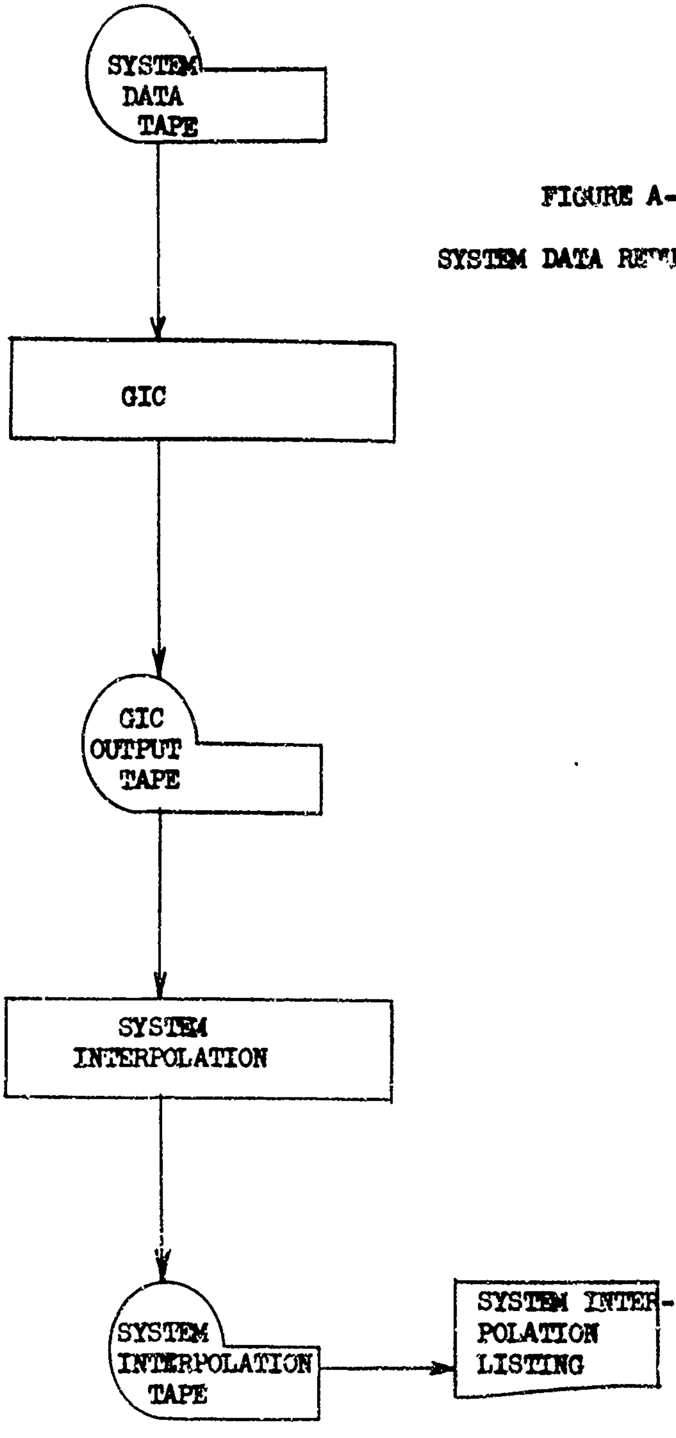


FIGURE A-2

SYSTEM DATA REDUCTION

Words consist of the word sync pattern (WS), the position data and the word count (WC).

Example:

WS	WC	DATA + PARITY

bit 1 . . . . 12 13 . . . . 16 17 36

Word sync (first 6 to 12 bits of the word) should consist of at least 6 and up to 12 bits.

Suggested Pattern: 1001 1001 1001

Word Count. Every word in the frame should be identified by a number, i.e., 1-15, so that in case synchronization is lost, most of the shifted words can be recovered by computer program.

Data. Either the most significant or least significant bit may occur first. Parity may be odd or even.

Word Length. Maximum word length is determined by:

$$\underbrace{WS_1 + WC + DATA}_{WORD_1} + \underbrace{WS_2}_{\text{WORD SYNC OF } WORD_2} \leq 48 \text{ bits}$$

In case a proposed system has gaps between words, the gaps should be constant and filled with spacer bits (all zeros). The maximum word length is then determined by:

$$\underbrace{WS_1 + WC + DATA + \text{Spacer bits}}_{WORD_1} + \underbrace{WS_2}_{\text{WORD SYNC OF } WORD_2} \leq 48 \text{ bits}$$

(2) Recording (see Appendix D, Instrumentation)

FM Recording (preferred)

Recording Mode: Level Changes on "1"
Manchester Code
Bipolar

Direct Recording

Recording Mode: Manchester Code

Recording of the computer clock is desirable.

The words which the system computer should output and the desired order of these words is:

System Time
System Latitude
System Longitude
System Altitude
System Wander Angle (if used)
System X Velocity or North Velocity
System Y Velocity or East Velocity
System Z Velocity or Vertical Velocity

These words are required information for all systems. In addition, special system configurations (stellar tracker, dopplers, etc.) will require additional output words. These should be coordinated in advance with CIGTF personnel.

3.5.4 If the navigation system output format meets the above specifications, the GIC output format can be standardized as follows:

One Data Word = 1 Computer Word

One Record = 1 Frame

IRIG TIME WORD = 48 bits

Word Structure:

Data Word

MARKERS	WORDCOUNT	DATA
47 - 42	41 - 36	35 . . . 15 0

Time Word

1	0 0	← SFC →	0	0	← MS →
47 46	45 29	28 12	11	10	9

In case of long test flights (long input tape) the GIC will produce two or more output tapes without interrupting the conversion process.

3.6 Error Data Reduction

3.6.1 System error information is obtained by computing the difference between system indicated data and reference standard data at ten second intervals. Radar time is derived from a master range timing station through land line and microwave relay. For transport flights the system timing reference is recorded directly from an on-board IRIG-B format time code generator. A secondary system time for fighter flights is derived from a precision oscillator signal which is initially referenced to IRIG-B time. Using these methods, a time correlation within 1 millisecond between radar and system data is achieved in both aircraft.

3.6.2 Figure A-3 shows the data reduction flow chart for system error information. The Position Error program assures that the system time series and radar time series are identical. It calculates the differences in the system and reference quantities and outputs these differences at ten second intervals. These error results are input into the Del-Marge program which combines on a single magnetic tape a time series of the system error data for all flights. The Analyze program computes the following quantities at ten second intervals for the total ensemble of flights:

Mean Latitude Error ($\overline{\Delta X}$)

Mean Longitude Error ($\overline{\Delta Y}$)

Mean Radial Error ($\overline{\Delta R}$)

Standard Deviation of Latitude Error (σ_x)

Standard Deviation of Longitude Error (σ_y)

Standard Deviation of Radial Error (σ_r)

Median of Radial Error

This program also computes an estimate of the distribution function of the radial error and calculates percentile errors for each time point. Plots of these percentiles are the primary accuracy results of a test program.

3.6.3 An additional method is sometimes used for obtaining velocity error data. The Smooth program will fit a quadratic or cubic to the position error curve (results of the Position Error program) and differentiate the curve to obtain velocity errors. These errors are output at one minute intervals. The Del-Merge and Analyze programs then can be used to accumulate velocity error information exactly as they did for position error information.

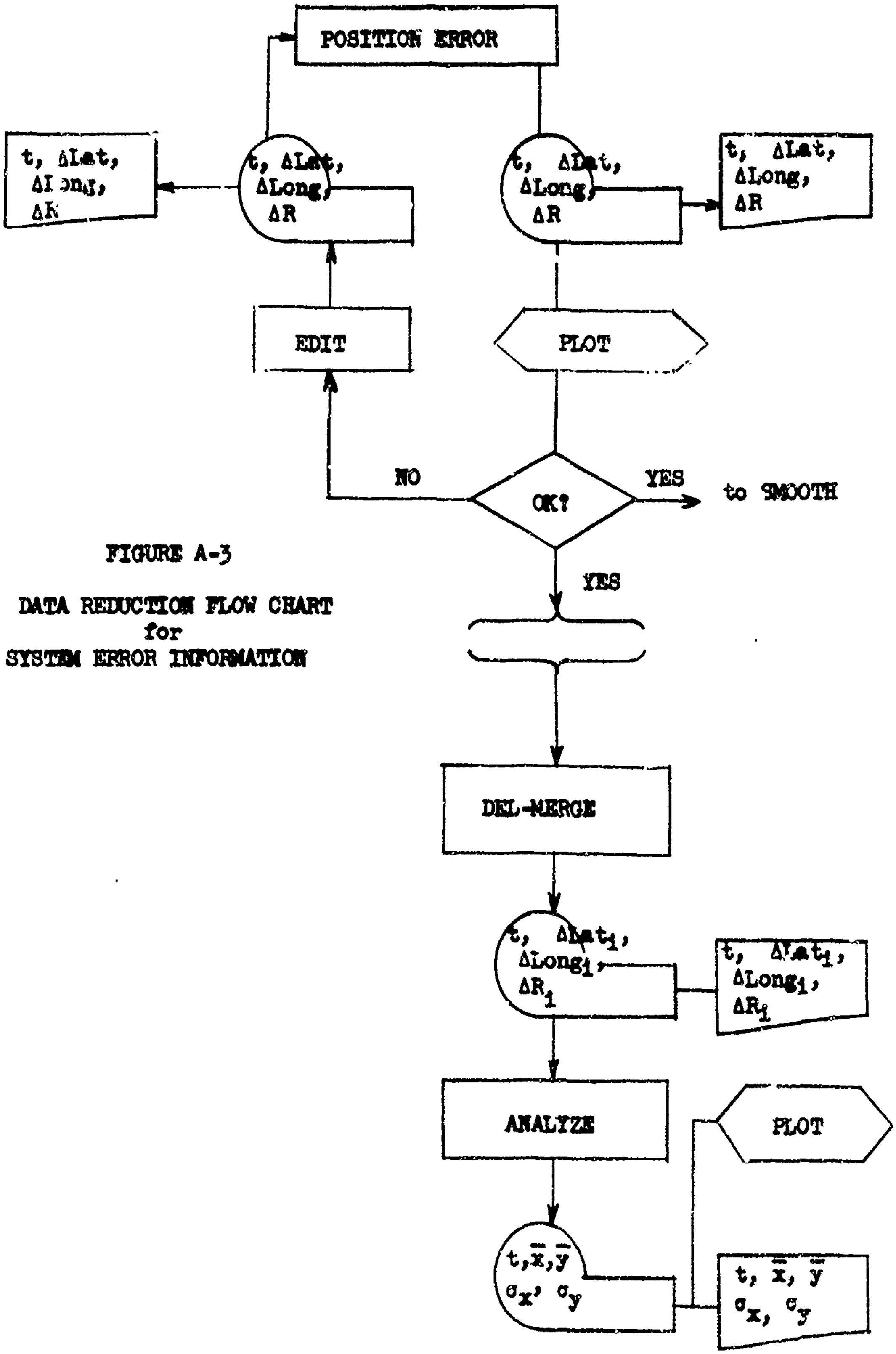


FIGURE A-3

DATA REDUCTION FLOW CHART
for
SYSTEM ERROR INFORMATION

4. CALCULATION OF SYSTEM ACCURACIES

4.1 The cumulative error plots of latitude error and longitude error versus time for the ensemble of test flights are the basis for system accuracy calculations. These plots are cross-sectioned at every point in time (one minute intervals). The means and standard deviations of the distribution of errors at each time are calculated. To obtain the 50th (CEP) and 90th percentile curves, a weighted sum of non-central chi-squares is approximated by fitting its first two moments to those of the ordinary central chi-square, χ^2. The Wilson-Hilferty transformation is then used to transform χ^2 to a normal variable. The resulting percentile value, ΔR_γ is given by:

$$\Delta R_\gamma = \sigma\sqrt{m}\left[1 + K_\gamma\left(\frac{v}{9m^2}\right)^{1/2} - \left(\frac{v}{9m^2}\right)\right]^{3/2}$$

where

$$m \text{ (mean)} = 1 + \sum_{i=1}^{2}\left(\frac{\bar{x}_i}{\sigma}\right)^2$$

x_i is a coordinate of miss distance

$$v \text{ (variance)} = 2\left[\sum_{i=1}^{2}\left(\frac{\sigma_{x_i}}{\sigma}\right)^2 + 2\sum_{i=1}^{2}\left(\frac{\sigma_{x_i}}{\sigma}\right)\left(\frac{x_i}{\sigma}\right)^2\right]$$

K_γ is obtained from a table of the cumulative normal integral ($K_{50\%} = 0$, $K_{90\%} = 1.281562$)

The use of this approximate method has been checked by actually calculating percentiles of the theoretical distribution of the radial error for each point in time and comparing the results. In all cases for $.5 \le \Delta R \le .9$, an error of less than 1% was noted. The approximate method is therefore used because it saves considerably on digital computer time.

4.2 Another method has been developed whereby the distribution is modeled and calculated from a fit of the test data points. This method is projected for possible use in future test programs.

5. THEORY AND APPROACH TO ERROR COEFFICIENT RECOVERY (Supplementary Test Results)

The problems associated with the testing of inertial navigation systems were first considered at the CIGTF in 1963. The many years of experience in obtaining quantitative results from the laboratory and sled testing of inertial components emphasized the desirability of obtaining more than an error plot from flight test results. The problems of determining error coefficients

from inertial navigation systems seemed insurmountable; not only do a multitude of error sources exist, but many of their propagations are also so highly correlated that the sources appear inseparable analytically. The equations which have been formulated and what are the basis of the CIGTF error analysis are:

$$\ddot{\overline{\delta R}} + 2W_{IT} \times \dot{\overline{\delta R}} + \dot{\overline{W}}_{ET} \times \overline{\delta R} +$$

$$+ [(\overline{W}_{IT} + \overline{W}_{IE}) \cdot \overline{\delta R}] \; \overline{W}_{ET} +$$

$$+ (W_S{}^2 + W_{IE}{}^2 - W_{IT}{}^2) \; \overline{\delta R} =$$

$$= \overline{\delta K}_o - \overline{A}_{CP} \times \overline{a}_P + (\overline{M}) \; \overline{a}_P - 2\delta W_S \; (W_S) \; \overline{R}_C$$

and $$\dot{\overline{A}}_{CP} + \overline{W}_{IT} \times \overline{A}_{CP} = \overline{\varepsilon}$$

where

$\overline{\delta R}$ = position error vector

$\overline{W}_{IT}$ = $\overline{W}_{IE} + \overline{W}_{ET}$ = rotation rate vector of the true reference with respect to the inertial frame

$\overline{W}_{ET}$ = rotation rate factor of the true reference frame with respect to the earth's frame

$\overline{W}_{IE}$ = earth's rate vector

$W_S{}^2$ = g/R = Schuler frequency

$\overline{A}_{CP}$ = vector angle from computer frame to platform frame

$\overline{a}_P$ = true platform acceleration vector

δW_S = error in computer Schuler frequency

$\overline{R}_C$ = earth's radius of curvature vector

$\overline{\varepsilon}$ = platform drift rate vector caused by gyro drift rates and torquer scale factor errors

$\overline{M}$ = matrix of scale factors and misalignments

$\overline{\delta K}_o$ = accelerometer bias error vector.

These are linear differential equations with time varying coefficients which assume that the airborne computer mechanization of the pure inertial mode of operation is perfect. Together, they represent the error model for a system which would operate perfectly if the initial conditions were correct and if there were no component errors.

Since the equations are linear, the effect of all individual error sources is additive. By driving the equations with one nominal error source at a time, individual error coordinate functions may be generated. These coordinate functions may then be used with the measured system error to solve for the error coefficients in a least-squares sense. These coefficients are universal parameters of system performance and can be used to simulate the system error propagation for any specified flight profile.

With this groundwork laid, the testing of inertial navigators began in February 1964 with a test program to evaluate the Autonetics XN-16 system. It was found that the analytical approach as postulated was inadequate in three areas:

(1) The error model (driving coordinate functions) was incomplete; for example, it did not include the effects of gyro and accelerometer misalignments, and these were found to be significant sources of error. After an intensive study was performed to determine possible sources of error which had been omitted, the error model was expanded from 19 to 30 terms.

(2) Because of the multiple correlation of the error functions, the least squares matrix was invariably singular. An analysis of the multiple correlation of these error functions revealed a method of grouping highly correlated error functions together in the least squares matrix and solving for the error coefficient of the grouping. A computer program was written to do this analysis.

(3) Even after these problems had been countered, the solution for error coefficients was found to be ineffective because the uncertainties in the many error sources once again caused a weak solution to the least squares matrix. It was necessary to insert *a priori* estimates of the standard deviations of the error coefficients obtained from calibrations to obtain meaningful error coefficients from the program. To combat the obvious possibility that the inserted *a priori* estimates were forcing the solution, a "Figure of Test Merit" was devised to indicate the extent to which the inserted estimate influenced the solution.

These changes to the original methods now allow the extraction of meaningful error coefficients from navigation systems. These powerful analytical methods are currently being used with great success on the Stellar Inertial Doppler System to determine the fundamental operation of the inertial reference unit.

Every system tested has its own idiosyncracies which must be accounted for in the evaluation. As systems become more sophisticated, their error models become more complex. For example, no computer error is assumed in the present models. As system accuracies increase, this effect will have to be modeled. This analytical approach represents the CIGTF method of providing the best possible equipment at the lowest possible cost to the man in the cockpit.

(Not Part of the Standardized Test)

APPENDIX B

LABORATORY TESTING

1. INTRODUCTION

The accuracy demanded in aircraft inertial navigation systems requires a complete, detailed evaluation of all the components of the system. The CIGTF has facilities available to test and provide meaningful laboratory evaluation of each component received for testing. This appendix provides test information on both gyroscopes and accelerometers. In addition, Section 4 and 5 define environmental system and star tracker tests.

2. GYROSCOPE TESTS

2.1 In order to acquire a high level of statistical confidence in the evaluation of a specific type of gyroscope, it is advantageous to test more than one gyroscope. A typical gyro test program will last between one and three months, assuming three specimens are available and are tested simultaneously.

2.2 Subsystem Concept. To make testing conform as closely as possible to actual conditions, the subsystem concept is employed. A gyro mount is fabricated to simulate the actual navigator mounting structure in terms of mass, heat transmissability, and physical location of components. The navigator heater blankets and temperature controller are used to control mount temperature. In addition, where practical, excitation electronics identical to those to be used in the aircraft are used in testing.

2.3 Laboratory Tests. The following tests have been designed to investigate gyro performance in light of specific operational requirements of an inertial navigator. A single-degree-of-freedom gyro is assumed throughout; however, tests for a two-degree-of-freedom gyro are usually identical except for the additional orientations required for the two sensitive axes.

2.3.1 Preliminary Tests. Preliminary tests consist of all tests necessary to check out the gyro, the gyro electronics, and the mating of the gyro to its mount and to the test table.

2.3.2 Standard Torque-to-Balance (STB) Test. The standard torque-to-balance (STB) test is a tumbling test in which a rate-drive table is driven at a constant angular velocity such as twenty Earth rate. The gyro signal generator and torque generator are connected in the torque-to-balance mode. Sampling of the torque generator current provides data which yields the following information: drift coefficient magnitudes, wheel-on instabilities, and wheel-shutdown instabilities. This drift coefficient information can be used to computer compensate gyro drift in a navigator.

2.3.3 Non-Compensable Drift Test. This test is performed with the gyro connected in the servo mode so that the signal generator output controls rotation of the test table. The table axis and the sensitive gyro axis are both horizontal or both vertical; thus, usual navigator component orientations are simulated.

Compensation is applied for Earth rate and gyro drift. Then without any further adjustment of compensation, the gyro is allowed to drift for several hours. The drift rate measured after compensation is the non-compensable drift of the gyro which indicates the fixed position total drift rate wheel-on instability. This information could be used to establish optimum filter weights in a Kalman mechanization.

2.3.4 Sensitivity Test. The sensitivity test indicates how variations in gyro operating and environmental parameters affect fixed position total drift without compensation for Earth rate and gyro drift. Again, the gyro is oriented as it would be in a navigator.

The following parameters are varied one at a time above and below the normal values while the others are held at the normal value: wheel supply frequency, wheel supply voltage, gyro temperature, signal generator excitation voltage, external magnetic field, and gyro temperature gradients. The fixed position total drift rate is recorded at each parameter value and the results are usually displayed graphically.

2.3.5 Environmental Tests. A gyro is subjected to three types of environmental tests while non-operating. These tests simulate conditions that a gyro might undergo during shipment or between flights. The tests are: hot soak, cold soak, and mechanical shock. Immediately prior to and immediately after each environment an STB test is performed to measure any changes in the gyro drift coefficients caused by the environment.

A fourth environmental test, mechanical vibration, is performed with the gyro operating in order to simulate aircraft vibration. The drift coefficients are evaluated before and after vibration to measure the effect of the test.

2.3.6 Warmup Test. The purpose of this test is to determine the warmup characteristics of the gyro and, in particular, to determine the time required for the gyro to achieve stable operation after turn-on.

The gyro is connected in the torque-to-balance mode and oriented with one axis vertical. Fixed position total drift rate and gyro temperature are recorded as a function of time while the gyro is heated to normal operating temperature. This information can be used to compute a warmup time, or to computer compensate the gyro output during warmup.

2.3.7 Autocorrelation Test. The purpose of this test is to determine the autocorrelation function of the gyro fixed position total drift rate. The gyro is operated in the torque-to-balance mode with the spin axis vertical and the sensitive axis north. A compensation current is applied to hold the signal generator at its null position. After gyro temperature and drift rate have stabilized, the torque-to-balance current is sampled periodically. From this information the autocorrelation function can be computed.

Typically, this autocorrelation function plotted versus time takes the form of a decaying exponential. The time constant of such an exponential is defined as the autocorrelation time of the gyro. This value determines the amount of time necessary to predict, with a known confidence level, the mean value of the gyro drift rate.

2.3.8 Fixed Position Total Drift Rate and Torque Generator Scale Factor Test. This test gives information from which the fixed position total drift rate and torque generator scale factor magnitudes, wheel-on instabilities, and wheel-shutdown instabilities are obtained for navigator orientations.

With the gyro connected in the torque-to-balance mode and the spin axis vertical, the sensitive axis is directed alternately north and south while the torque-to-balance current is recorded. Repetition of the test with the wheel-on and then with wheel-shutdowns allows computation of the above quantities. The test is repeated with the spin axis horizontal and the sensitive axis vertical to obtain the fixed position total drift rate for the other gyro orientation used in navigators.

2.3.9 Torque Generator Scale Factor Long Term Instability Test. Since the torque generator scale factor is not frequently updated, it is important that this scale factor be stable. The standard deviation of all determinations of the torque generator scale factor during a test series is computed and defined as the scale factor long term instability.

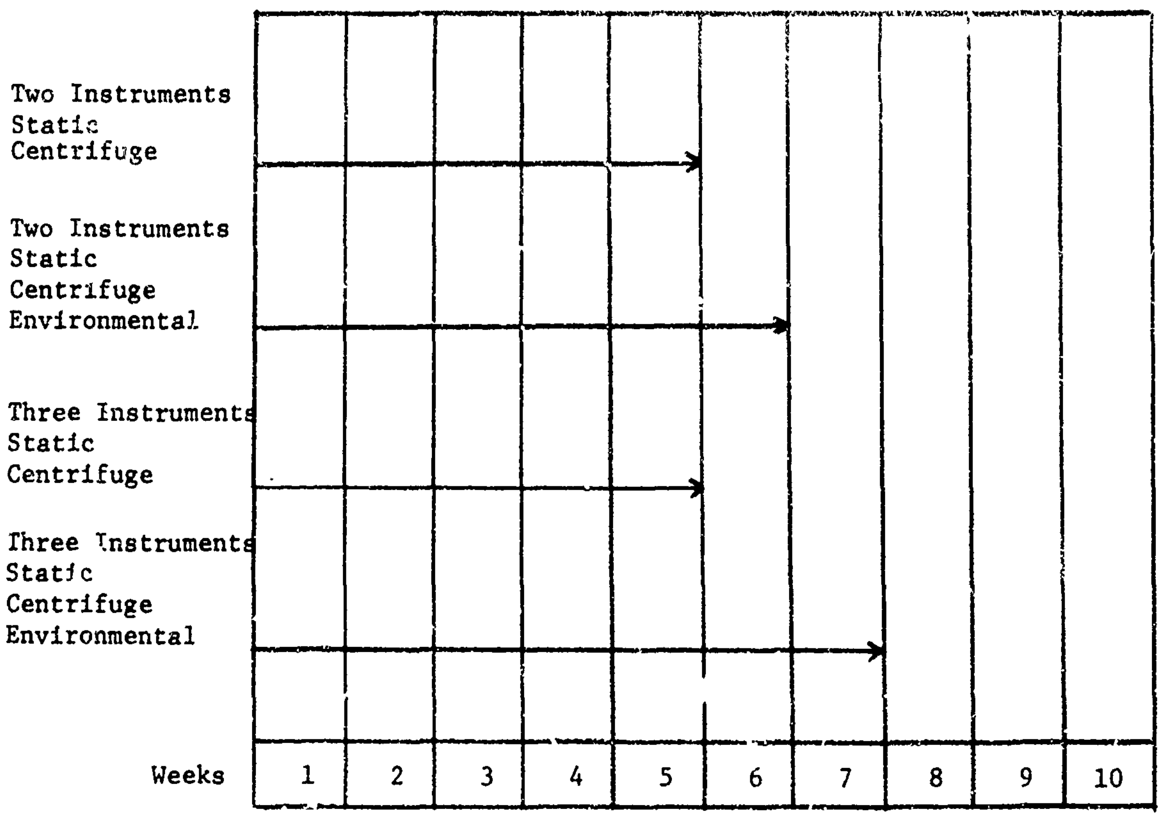
3. ACCELEROMETER TESTS

3.1 In order to acquire a high level of statistical confidence in the evaluation of a specific type of accelerometer, it is advantageous to test from two to three accelerometer specimens. A typical accelerometer test program, as described below, will last between one and two months (see Figure B-1).

3.2 Static Testing. Static testing consists of conducting the following tests in a 1 g environment.

3.2.1 Initial Checkout. The initial checkout consists of a visual check for damage in shipment, a continuity check for any open or shorted electrical circuits in the instrument, and an operational check where power is supplied to the instrument and the output is monitored.

3.2.2 Input Axis Alignment. The accelerometer is mounted on a dividing head with its input axis nominally in the dividing head plane of rotation and its output axis nominally perpendicular to that plane. In this configuration, the input axis is constrained to rotate in the local gravity field. The dividing head is then rotated 180° $\pm$ 0.3 arc seconds and the output again recorded. The dividing head position is then adjusted until the accelerometer output equals the average of the above two recorded outputs. The above sequence is repeated until equal outputs are obtained, indicating that the input axis is horizontal. The final position with the input axis horizontal and the pendulous axis directed down is noted as the 0° reference position, and the dividing head angle is noted as the reference angle.



TESTING TIME

Figure B-1

3.2.3 Two-Point Test. The accelerometer is mounted on the dividing head in the same position as for the input axis alignment sequence and an input axis alignment is performed. The accelerometer output is then recorded with the input axis positioned alternately at the 90° and 270° reference positions (corresponding to +1 and -1 g acceleration inputs, respectively). Twenty rotations are performed per test.

3.2.4 Twelve-Point Linearity Test. The accelerometer is mounted in the same configuration as for the two-point test. Starting at the 0° reference position, data are taken at 30° increments. A complete test consists of twenty rotations of the dividing head, alternately rotating clockwise and counterclockwise. During the test, the dividing head position is repeated to within 0.3 arc seconds.

3.2.5 Threshold and Resolution. For the threshold test, the input axis is positioned at the 0° reference. The dividing head is first rotated counterclockwise ten arc seconds, and clockwise twenty arc seconds, then counterclockwise ten arc seconds, returning to the initial position. At each 0.5 arc second increment during the above rotations, the accelerometer output is recorded.

For the resolution test, the above procedure is repeated except that the reference position is with the instrument's input axis 60° above the 0° reference. The accelerometer output is recorded at one arc second increments instead of 0.5 arc seconds.

3.2.6 Parameter Variation. In the parameter variation tests, the twelve-point test procedures are followed with one of the input voltages or frequencies to the accelerometer varied 10% of the nominal value in four equal increments above and below the nominal operating value.

3.3 Centrifuge Testing

3.3.1 Placing an accelerometer on a centrifuge is the most economical way of subjecting an instrument to sustained acceleration above 1 g. Also, by accurately controlling the rotation of the centrifuge arm, very precise readings of the accelerometer output can be obtained.

3.3.2 The accelerometer output is recorded over a 20 g range in increments of 1 g to determine departure from linearity. However, the centrifuge has a 25 g range with an infinite number of steps to 25 g's.

3.4 Environmental Testing. Environmental tests are accomplished to determine if the accelerometer can operate correctly after being subjected to established extremes in temperature, vibration, and mechanical shock. The test extremes are set by military specifications as follows:

3.4.1 Hot Soak, MIL-E-5272C, Para 4.1.2.

3.4.2 Cold Soak, MIL-E-5272C, Para 4.2.2.

3.4.3 Vibration, MIL-T-5422E, Para 4.2.1, Part II.

3.4.4 Mechanical Shock, MIT-T-5422E, Para 4.3.2.1.

4. SYSTEM ENVIRONMENTAL TESTS

4.1 These tests determine t' sensitivity of a system to selected environmental factors. Tests are cond .ted with the system in both operating and non-operating conditions. Calibrations are performed prior to and after environmental testing, and between individual tests. Performance degradation is determined by comparison of these calibrations and by comparison of position and velocity error plots made during navigation runs in operating condition tests.

4.2 Because the environmental specifications to which systems are designed vary, these test outlines include only the ranges of environmental conditions which can be achieved. Tests will be tailored to meet specific equipment design specifications.

4.2.1 Temperature Variation (Non-Operating System)

(1) Low Temperature. The entire system is placed in an environmental chamber and the temperature reduced to the specific level. After thermal stabilization, the temperature is returned to room ambient. System warm-up time is recorded and plotted.

(2) High Temperature. The entire system is placed in an environmental chamber and the temperature increased to the specified level. After thermal stabilization, the temperature is returned to room ambient. System cool-down time is recorded and plotted.

(3) Maximum Temperature Variations: -100°F to +200°F.

4.2.2 Temperature-Altitude Simulations (System Operating)

(1) Low Temperature. With the system operating in an environmental chamber, the temperature is reduced to the specified level. Pressure is then reduced to the equivalent of the specified altitude. After thermal stabilization, system performance is monitored during a bench navigation run. The temperature and pressure are returned to room ambient.

(2) High Temperature. With the system operating in an environmental chamber, the temperature is increased to a specified level. After thermal stabilization, system performance is monitored during a bench navigation run. The temperature is then returned to room ambient.

(3) Maximum Variations: Temperature, -100°F to +200°F, Altitude, 0 to 220,000 feet.

4.2.3 Vibration Tests

(1) Magnetic Shaker. This test determines the effect of linear vibration on the system in both operating and non-operating conditions.

The tests are performed with one major unit of the system at a time on the vibration table.

In the operating condition, the major unit being vibrated is connected and operated with the remainder of the system.

Prior to this test, a sweep is made at a reduced vibration level to identify critical resonance frequencies.

Vibration Capacity: 0 to 5,000 pounds force

(2) Angular Vibration. This test is designed to evaluate the response of the system to simulated low altitude flight conditions. The system is operated on the Controlled Platform Test Stand which produces angular vibration about three axes simultaneously.

Frequency Range: 1/2 to 21 cps

Amplitude of Vibration: $\pm 4°$

Phase and amplitude of vibration about each axis are independently adjustable.

4.2.4 Mechanical Shock Test. This test is designed to evaluate the ability of the system to withstand mechanical shock, and is performed by arresting major units after a specified period of free fall. Shock is applied along specified axes of the units.

(1) Shock Pulse Shape: Half sine wave

(2) Duration: 11 $\pm$ 1 milliseconds

(3) Capacity: 800 pounds - 12 g maximum
25 pounds - 200 g maximum

5. STAR TRACKER TESTS

5.1 Standard star tracker tests are listed below in terms of the capabilities of the CIGTF Stellar Simulator. The simulator consists essentially of a fixed Dual Star Simulator (DSS) and a movable Single Star Simulator (SSS).

5.1.1 Spectral Response. This test determines the electro-optical sensitivity of the sensor to energy contained within defined wave length bands.

(1) The Stellar Simulator provides radiation between 0.35 and 1.0 micron wave lengths at 0.02 micron increments.

(2) To perform the test, the simulator intensity is set at a calibrated level and relative output of the sensor is plotted versus wave length as the simulator wave length is varied.

5.1.2 Window Refraction. The refractive properties of the window (housing) are determined by repeating the spectral response test for different orientations of the sensor line of sight with respect to the position of the simulated star.

5.1.3 Sky Background Polarization. This test determines the effect on performance of noise due to sky background polarization. The simulator can simulate a star on a sky background polarized between 0° and 180°.

5.1.4 Sensitivity to Star Fluctuation. Sensitivity to "twinkling" is measured by plotting sensor response against the frequency of modulation of star intensity. This modulation frequency is variable between 0 and 100 cps.

5.1.5 Star Magnitude Discrimination. The simulator can simulate two stars of variable magnitude and separation. Magnitude discrimination is evaluated by positioning two stars within the sensor's search field. Star magnitude is variable between -2.0 and +5.0 VM. The magnitude of one star is set at a programmed value which the sensor is commanded to seek. The magnitude of the other star is then adjusted until the sensor is unable to detect the difference in magnitude.

5.1.6 Star Magnitude Versus Background Tracking Ability. This test is performed by positioning stars of various magnitudes against sky backgrounds of various intensities. The star brightness is then decreased until the system can no longer acquire and track the star. A plot is then made of star magnitude versus sky brightness at which the system fails to track the star.

5.1.7 Sky Gradient Rejection Capability. The sky background of the DSS can simulate brightness gradients of 0, 2.5, 5.0, 10.0 and 20.0 percent per degree. The system is evaluated on its ability to sense and subtract sky gradient by requiring it to acquire and track stars against various gradients.

5.1.8 Search Rate. Search rates are measured by plotting tracker angle encoder angle versus time for various star magnitudes.

5.1.9 Mechanical Pointing Resolution. This test determines the minimum star displacement that can be detected by the star tracker. It is performed by allowing the tracker to acquire a stationary star and then displacing the star in one arc second increments along elevation and azimuth axes until the tracker realigns itself.

5.1.10 Tracker Pointing Accuracy. This test determines the readout accuracy of the azimuth and elevation angle encoders. The test is performed by recording and plotting encoder output versus position of a simulated star.

5.1.11 Field of View Size. This test is designed to evaluate the field of view of the tracker through the system (platform) housing. A star is positioned near the edge of the assumed field of view. The elevation angle is then reduced until the star is no longer detected by the tracker. This procedure is repeated at 30° increments through a 360° azimuth rotation of the tracker, and a polar plot made. A similar procedure is followed to determine the field of view at the upper elevation angle limit of the tracker.

5.1.12 Telescope Line of Sight (LOS) Stability. The tracker is positioned with the LOS collinear with the optical path to a simulated star. Angle encoder output is monitored during warm-up and changes in the ambient environment.

5.1.13 Misalignment of Star Tracker Reference Frame to System (IMU) Coordinates. The platform gimbals are locked with X and Y accelerometers horizontal. The tracker LOS is then aligned with a simulated star. Encoder outputs are recorded as the platform is rotated through 360° in 30° increments. The turntable on which the platform is mounted can be positioned to an accuracy of better than one arc second.

5.1.14 Double Star Detection. The DDS can simulate two stars of different magnitudes from superposition to a separation of 4°. This test evaluates the ability of the star tracker to detect, acquire, and track dual stars of varying magnitudes and separation.

APPENDIX C

FLIGHT PATTERNS

The flight patterns and tables included in this appendix are for Phase II (C-130) and Phase III (F-106 and C-130) pure inertial missions conducted during a CIGTF test program. Specific Phase III-A helicopter patterns will be designed at a later date. In addition, special patterns for Phases II-B and III-B (pure inertial), as well as standard II-A and III-A patterns for aided inertial tests will also be formulated when required. C-130 flight paths IIA-1, 2, and 3 are Phase II-A flight paths. All others shown are Phase III-A (F-106 and C-130).

TABLE C-I

C-130 FLIGHT PATH IIA-1

LEG NO	ALTITUDE (Ftx1000)	MAGNETIC COURSE (Degrees)	END LATITUDE	END LONGITUDE	DISTANCE (NM)	TAS (Knots)	TIME (Min)	ELAPSED TIME (Hr+Min)
1	30	Spiral Climb	32°52'N	106°06'W	0	200	20	0+20
2	30	302	35°10'N	108°54'W	196	280	42	1+02
3	30	122	32°52'N	106°06'W	196	280	42	1+44
4	30	302	35°10'N	108°54'W	196	280	42	2+26
5	30	122	32°52'N	106°06'W	196	280	42	3+08
6	30	302	35°10'N	108°54'W	196	280	42	3+50
7	30	122	32°52'N	106°06'W	196	280	42	4+32

TOTAL DISTANCE 1176 NM

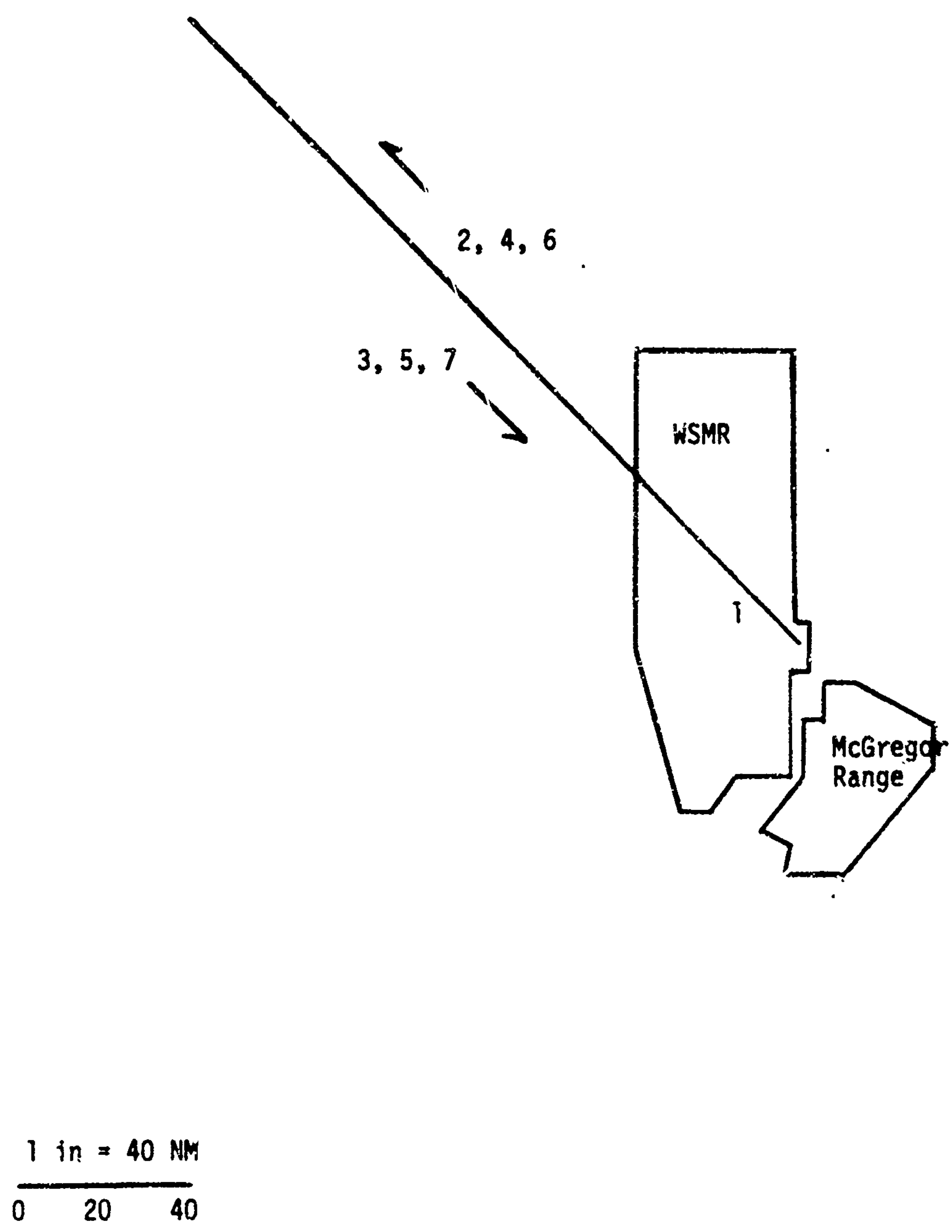
TABLE C-II

C-130 FLIGHT PATH IIA-2

LEG NO	ALTITUDE (Ftx1000)	MAGNETIC COURSE (Degrees)	END LATITUDE	END LONGITUDE	DISTANCE (NM)	TAS (Knots)	TIME (Min)	ELAPSED TIME (Hr+Min)
1	30	046	36°05'N	115°09'W	151	215	42	0+42
2	30	223	34°38'N	108°04'W	167	240	42	1+24
3	30	096	33°36'N	114°46'W	175	300	35	1+59
4	30	096	32°58'N	112°40'W	114	300	23	2+22
5	30	090	32°16'N	109°16'W	177	300	36	2+58
6	30	032	34°22'N	106°48'W	176	250	42	3+40
7	30	212	32°16'N	109°16'W	176	250	42	4+22
8	4.1	065	32°52'N	106°06'W	165	250	39	5+01

TOTAL DISTANCE 1301 NM

Fig C-1
C-130 FLIGHT PATH IIA-1



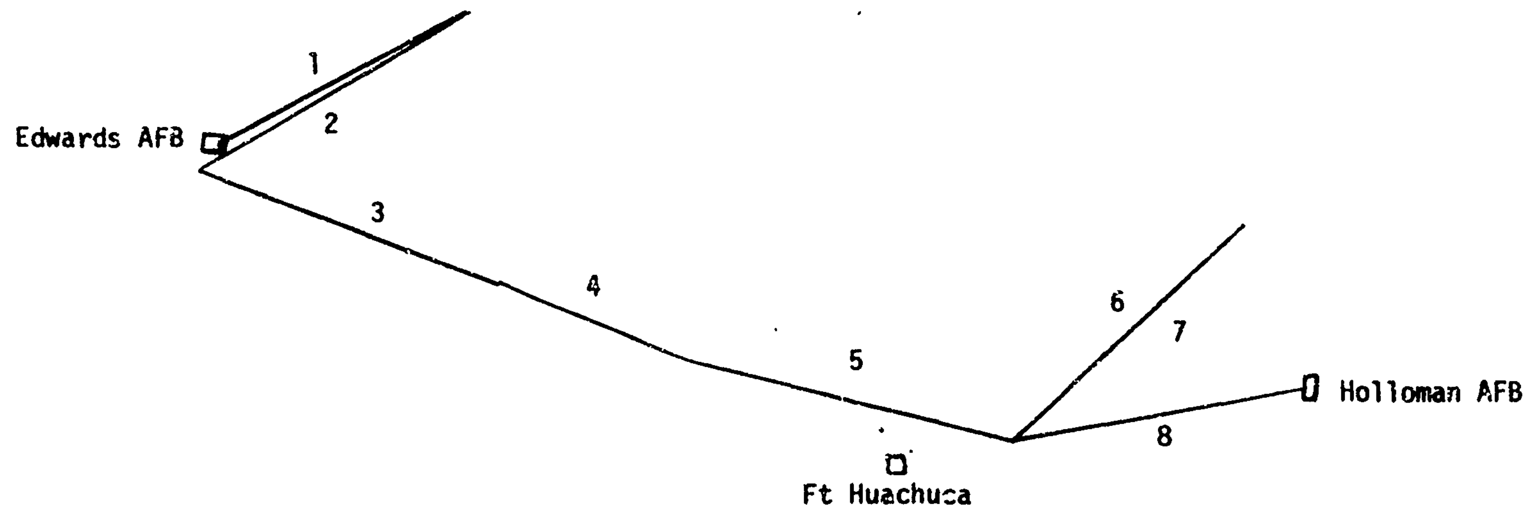


Fig C-2

C-130 FLIGHT PATH IIA-2

1 In = 100 NM

0 50 100

TABLE C-III

C-130 FLIGHT PATH IIA-3

LEG NO	ALTITUDE (Ftx1000)	MAGNETIC COURSE (Degrees)	END LATITUDE	END LONGITUDE	DISTANCE (NM)	TAS (Knots)	TIME (Min)	ELAPSED TIME (Hr+Min)
1	30	Spiral Climb	32°52'N	106°06'W	0	200	18	0+18
2	30	298	34°58'N	109°09'W	199	285	42	1+00
3	30	118	32°52'N	106°06'W	199	285	42	1+42
4	30	244	32°16'N	109°16'W	165	250	40	2+22
5	30	270	32°58'N	112°40'W	177	250	42	3+04
6	30	276	33°36'N	114°46'W	114	250	27	3+31
7	30	276	34°38'N	118°04'W	175	250	42	4+13
8	2.3	012	34°55'N	117°54'W	19	250	5	4+18

TOTAL DISTANCE 1048 NM

TABLE C-IV

F-106 FLIGHT PATH IIIA-1

LEG NO	ALTITUDE (Ftx1000)	MAGNETIC COURSE (Degrees)	END LATITUDE	END LONGITUDE	DISTANCE (NM)	TAS/MACH (Knots)	TIME (Min)	ELAPSED TIME (Hr+Min)
1	30	257	32°52'N	106°26'W	18	450	02	0+02
2	30	347	32°42'N	106°26'W	50	450	07	0+09
3	10 to 30	Various acrobatics in North Range area				450	75	1+24
4	30	153	32°52'N	106°06'W	53	450	07	1+31

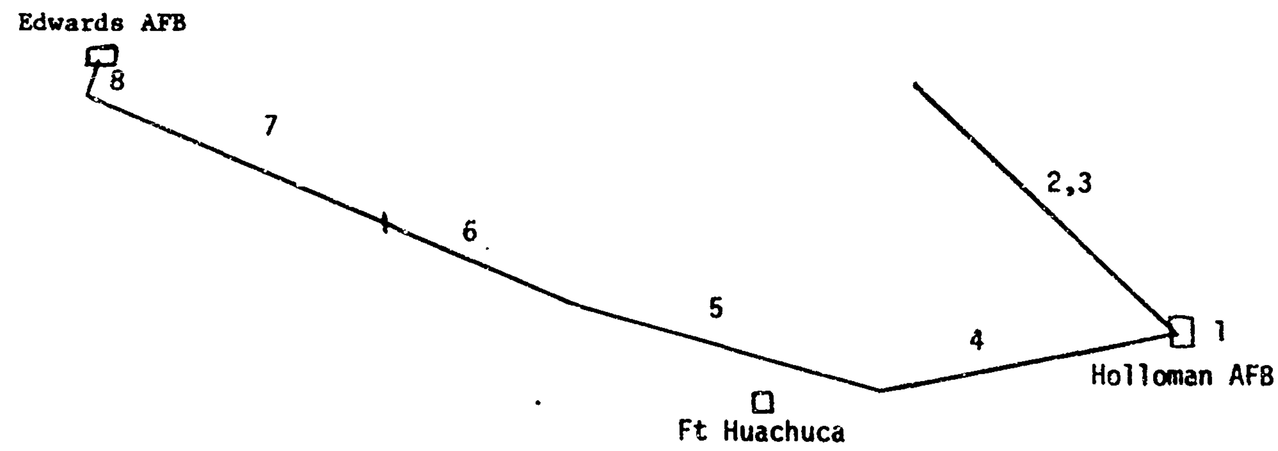
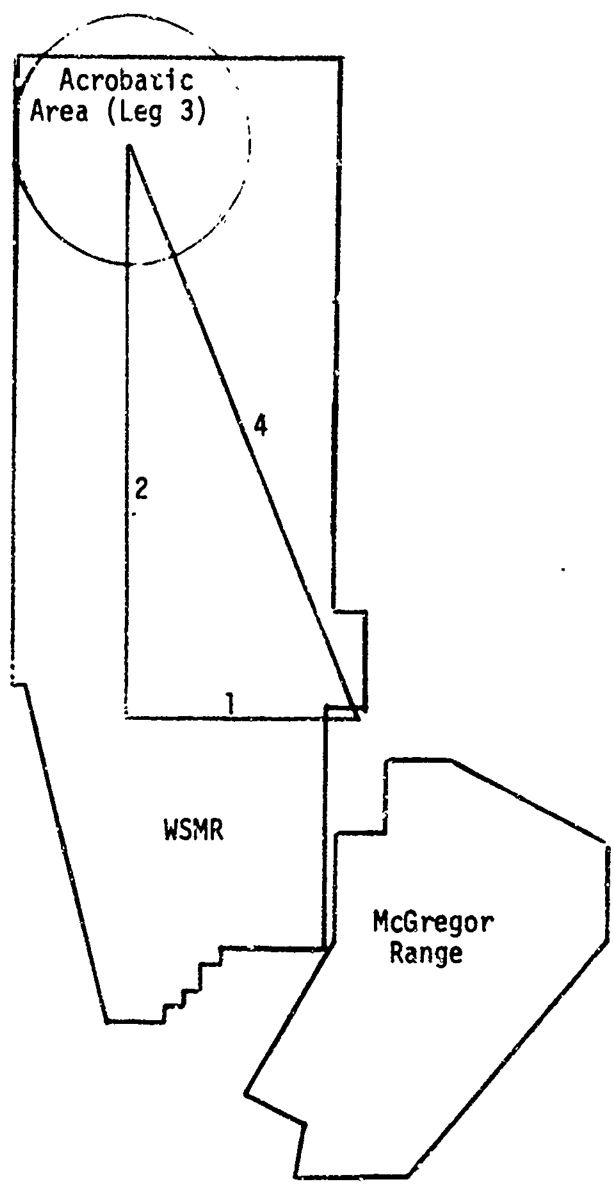


Fig C-3
C-130 FLIGHT PATH IIA-3

Fig C-4

F-106 FLIGHT PATH IIIA-1



1 In = 20 NM

0 10 20

TABLE C-V

F-106 FLIGHT PATH A-1

LEG NO	ALTITUDE (Ftx1000)	MAGNETIC COURSE (Degrees)	END LATITUDE	END LONGITUDE	DISTANCE (NM)	TAS/MACH (Knots)	TIME (Min)	ELAPSED TIME (Hr+Min)
1/9	40	244	32°21'N	108°42'W	136	515/0.9	16	0+16/1+37
2/10	40	352	34°20'N	108°30'W	120	515/0.9	14	0+30/1+51
3/11	40	020	35°09'N	107°51'W	58	515/0.9	07	0+37/1+58
4/12	40	058	35°41'N	105°57'W	99	515/0.9	12	0+49/2+10
5/13	40	085	35°36'N	105°13'W	36	515/0.9	04	0+53/2+14
6/14	40	133	34°57'N	104°42'W	47	515/0.9	15	0+58/2+19
7/15	40	169	32°49'N	104°44'W	128	515/0.9	15	1+13/2+34
8/16	40	260	32°52'N	106°06'W	69	515/0.9	08	1+21/2+42

TOTAL DISTANCE 1386 NM

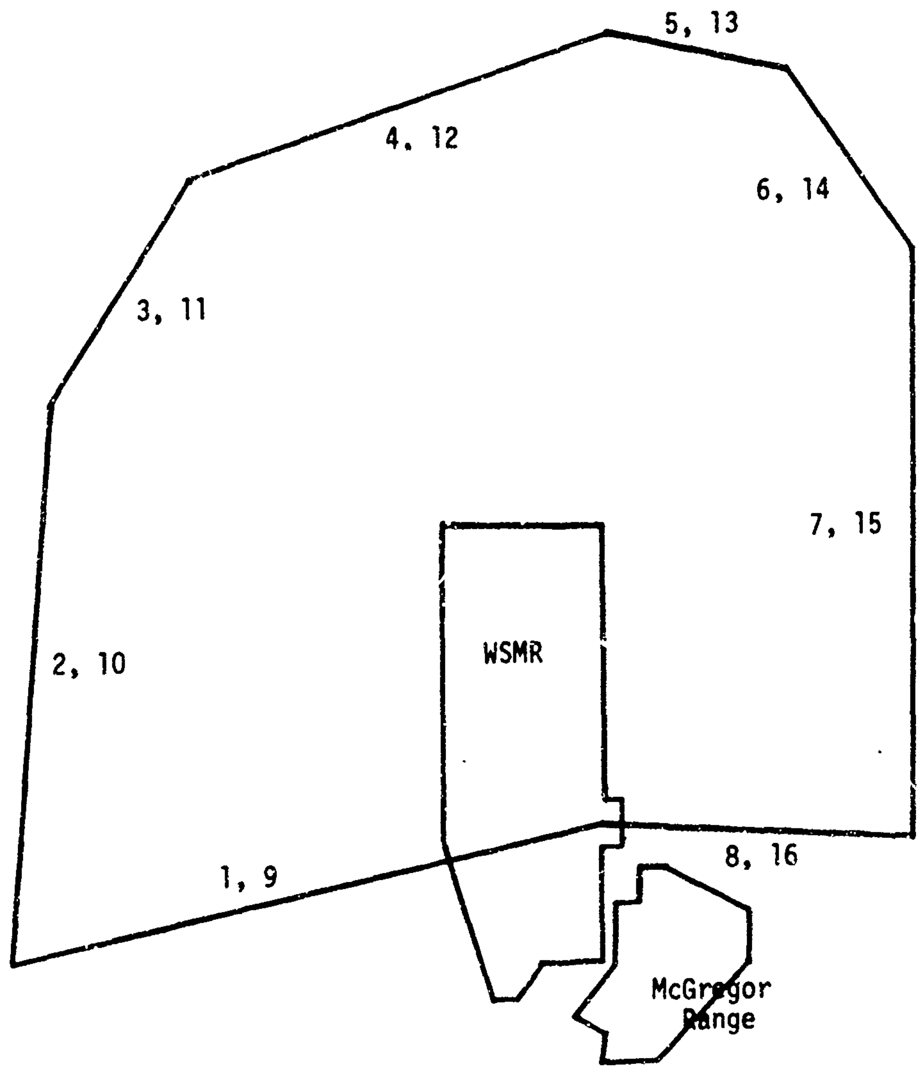


Fig C-5

F-106 FLIGHT PATH A-1

1 In = 40 NM

0 20 40

TABLE C-VI

F-106 FLIGHT PATH A-3

LEG NO	ALTITUDE (Ftx1000)	MAGNETIC COURSE (Degrees)	END LATITUDE	END LONGITUDE	DISTANCE (NM)	TAS (Knots)	TIME (Min)	ELAPSED TIME (Hr+Min)
1	*	347	34°07'N	106°09'W	75	450	10	0+10
2		212	33°45'N	106°37'W	30	450	04	0+14
3		137	33°24'N	106°22'W	23	450	03	0+17
4		167	32°34'N	106°22'W	50	450	07	0+24
5		257	32°34'N	106°27'W	5.5	450	01	0+25
6		334	33°08'N	106°36'W	36	450	05	0+30
7		023	34°01'N	105°50'W	64	450	09	0+39
8		227	33°33'N	106°49'W	55	450	07	0+46
9		047	34°01'N	105°50'W	55	450	07	0+53
10		203	33°08'N	106°36'W	64	450	09	1+02
11		154	32°34'N	106°27'W	36	450	05	1+07
12		077	32°34'N	106°22'W	5.5	450	01	1+08
13		347	33°24'N	106°22'W	50	450	07	1+15
14		317	33°45'N	106°37'W	23	450	03	1+18
15		032	34°07'N	106°09'W	30	450	04	1+22
16		167	32°52'N	106°09'W	75	450	10	1+32

TOTAL DISTANCE 677 NM

*Three profiles will be flown on this flight path at altitudes as follows:

- A-3/1 500' to 1000' terrain clearance
- A-3/2 Climb to 20,000' MSL, descend to 500' to 1000' terrain clearance, climb to 20,000 MSL
- A-3/3 Continuous climbs and descents between 500' to 1000' terrain clearance and 40,000' MSL

Fig C-6

F-106 FLIGHT PATH A-3

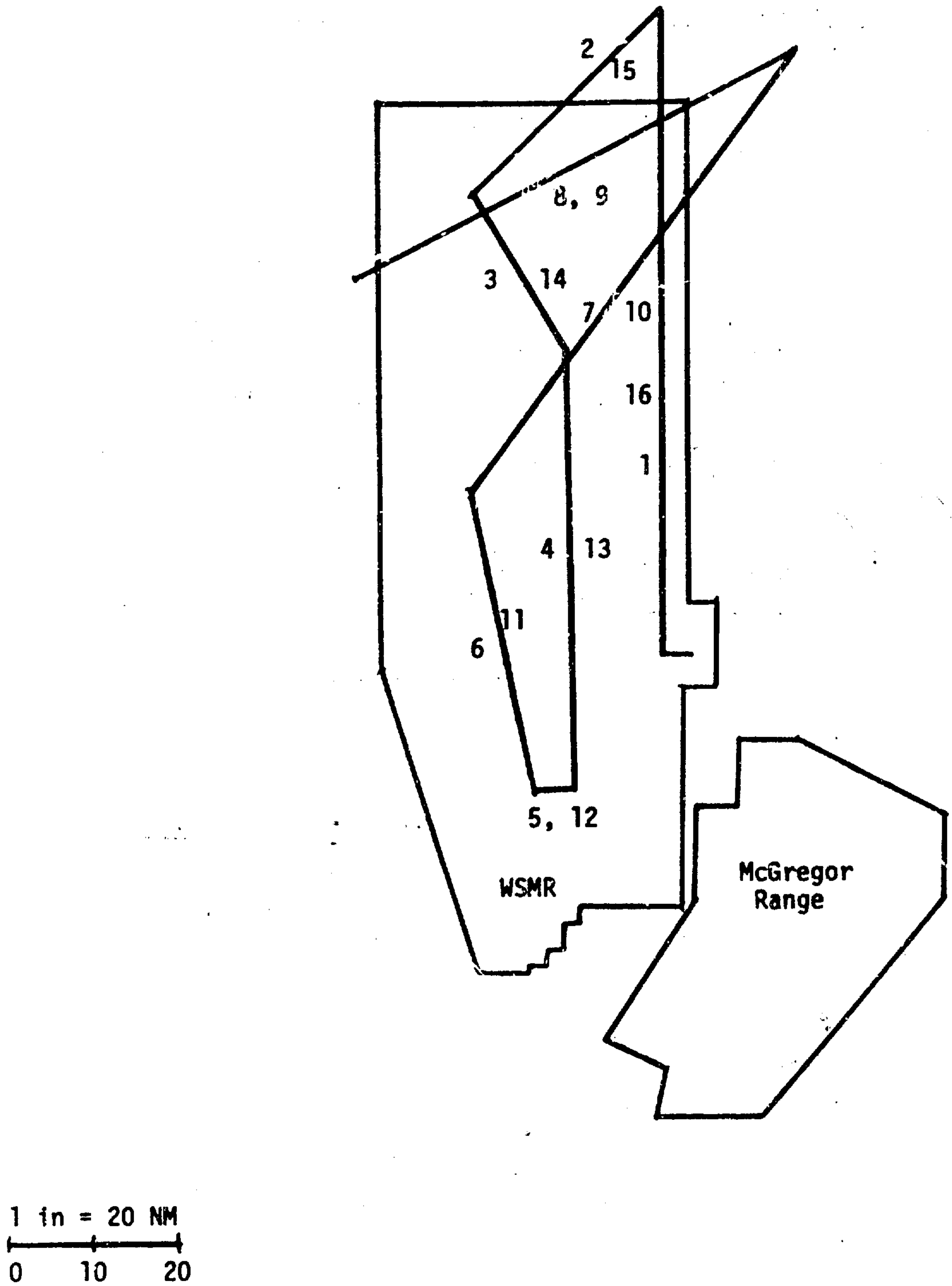


TABLE C-VII

F-106 FLIGHT PATH A-5

LEG NO	ALTITUDE (Ftx1000)	MAGNETIC COURSE (Degrees)	END LATITUDE	END LONGITUDE	DISTANCE (NM)	TAS/MACH (Knots)	TIME (Min)	ELAPSED TIME (Hr+Min)
1	40	347	34°13'N	106°09'W	80	0.9	09	0+09
2	40	257	34°13'W	106°24'W	11	0.95	02	0+11
3	40	167	32°45'N	106°24'W	89	Accel to 2.0	06	0+17
4	40	320	33°25'N	106°49'W	47	Decel to 0.9	05	0+22
5	40	034	34°02'N	106°03'W	53	515/0.9	06	0+28
6	40	257	34°02'N	106°50'W	38	515/0.9	04.5	0+32.5
7	40	126	33°22'N	106°07'W	53	515/0.9	06	0+38.5
8	40	211	32°54'N	106°41'W	48	515/0.9	05.5	0+44
9	40	077	32°52'N	106°06'W	33	515/0.9	04	0+48

TOTAL DISTANCE 452 NM

Fig C-7

F-106 FLIGHT PATH A-5

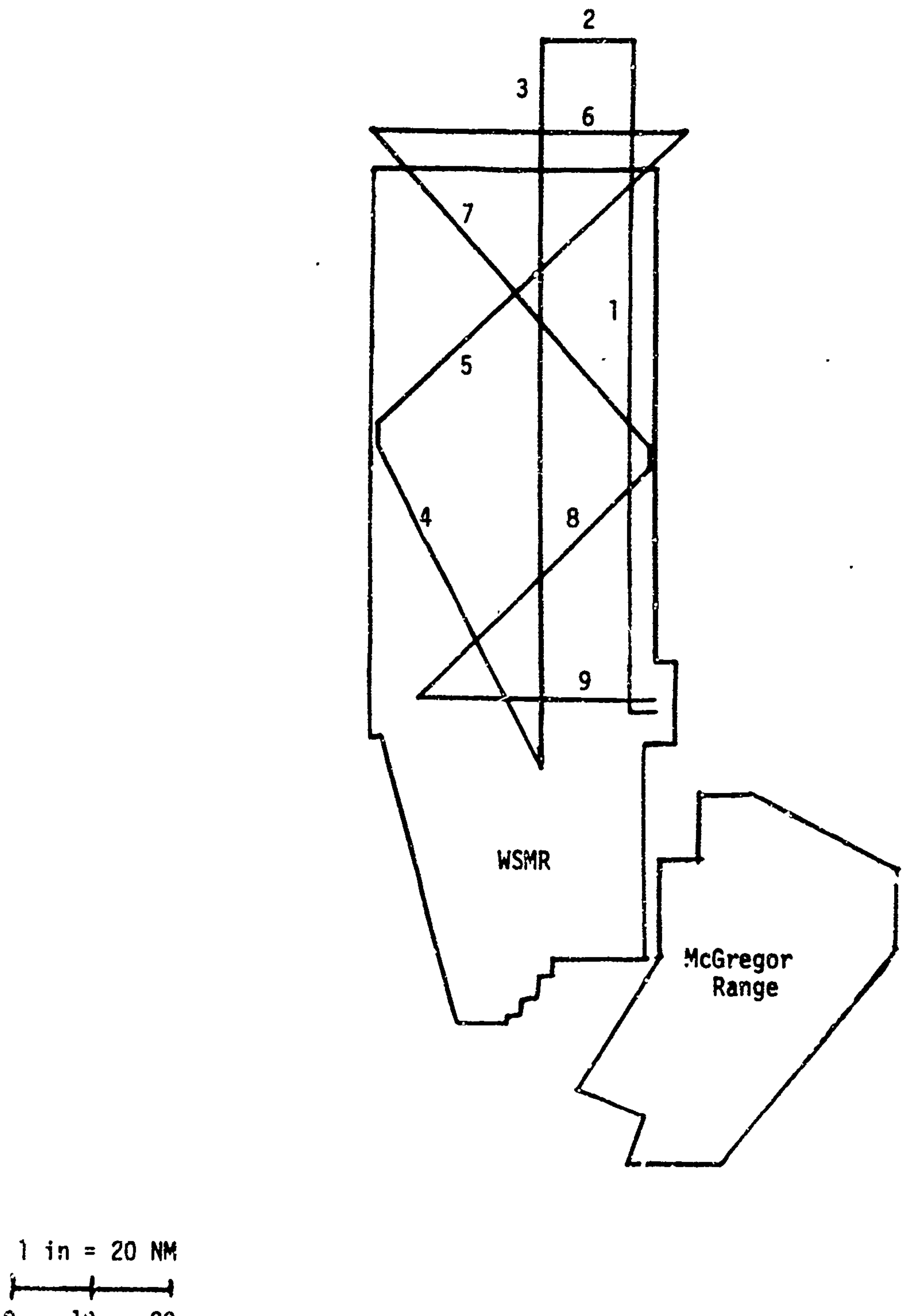


TABLE C-VIII

C-130 FLIGHT PATH A-1

LEG NO	ALTITUDE (Ftx1000)	MAGNETIC COURSE (Degrees)	END LATITUDE	END LONGITUDE	DISTANCE (NM)	TAS (Knots)	TIME (Min)	ELAPSED TIME (Hr+Min)
1	30	347	36°02'N	106°06'W	190	233	49	0+49
2	30	167	33°22'N	106°06'W	160	300	32	1+21
3	30	257	33°22'N	109°04'W	150	300	30	1+51
4	30	077	33°22'N	106°06'W	150	290	31	2+22
5	Descent	167	32°52'N	106°06'W	30	275	07	2+29

TOTAL DISTANCE 680 NM

TABLE C-IX

C-130 FLIGHT PATH A-2

LEG NO	ALTITUDE (Ftx1000)	MAGNETIC COURSE (Degrees)	END LATITUDE	END LONGITUDE	DISTANCE (NM)	TAS (Knots)	TIME (Min)	ELAPSED TIME (Hr+Min)
1	30	008	35°12'N	105°02'W	150	225	40	0+40
2	30	347	35°52'N	105°02'W	40	300	08	0+48
3	30	188	33°32'N	106°07'W	150	300	30	1+18
4	30	281	34°20'N	108°19'W	120	300	24	1+24
5	30	142	32°03'N	107°05'W	150	300	30	2+12
6	Descent	033	32°52'N	106°06'W	70	280	15	2+27

TOTAL DISTANCE 680 NM

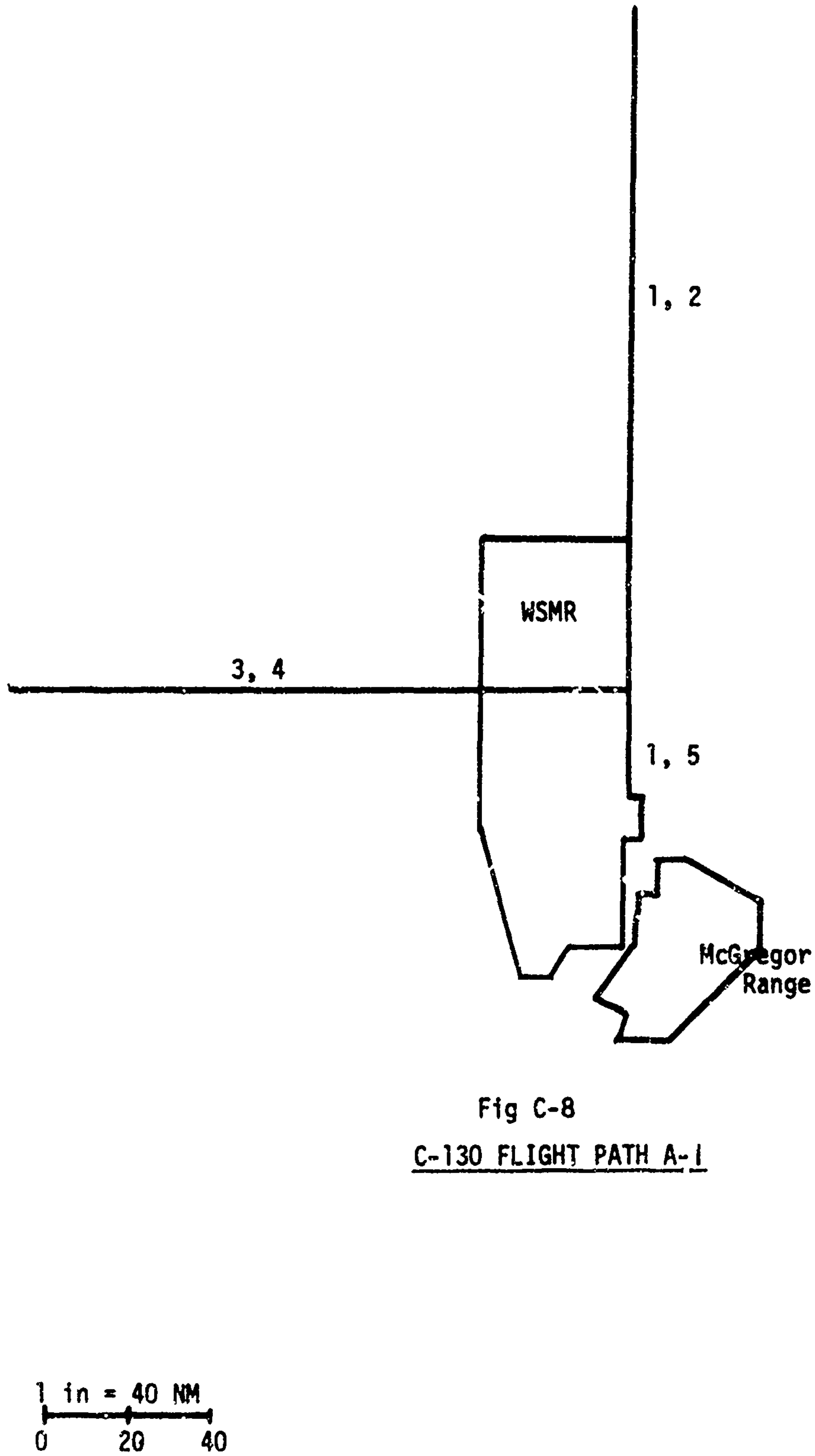


Fig C-8

C-130 FLIGHT PATH A-1

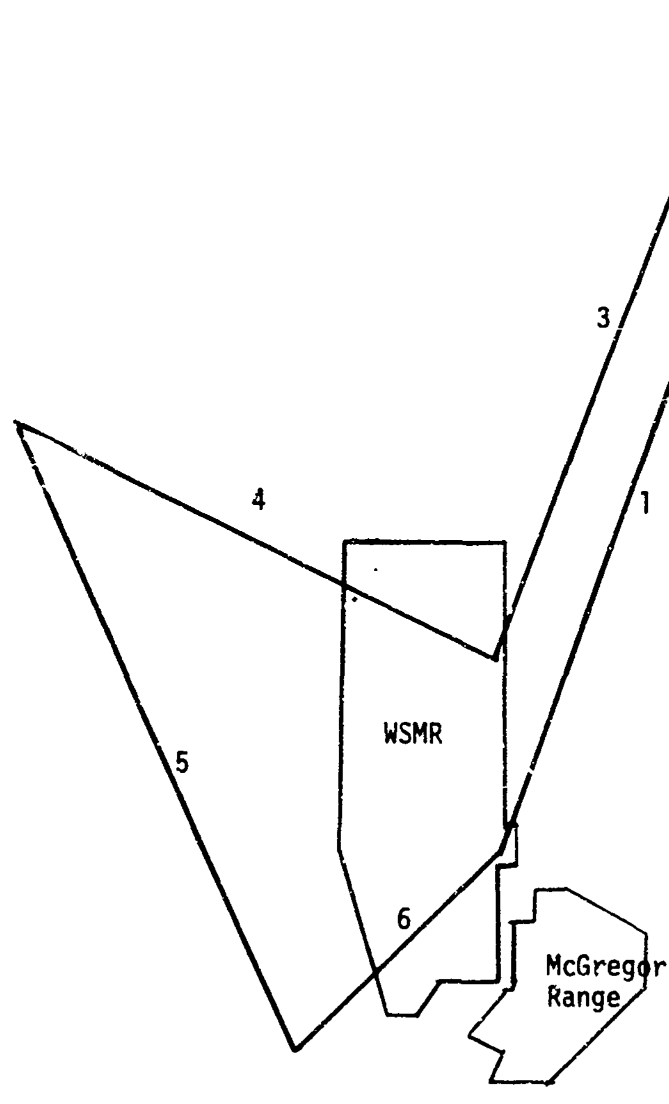


Fig C-9

C-130 FLIGHT PATH A-2

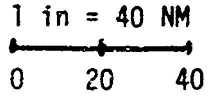


TABLE C-X

C-130 FLIGHT PATH A-3

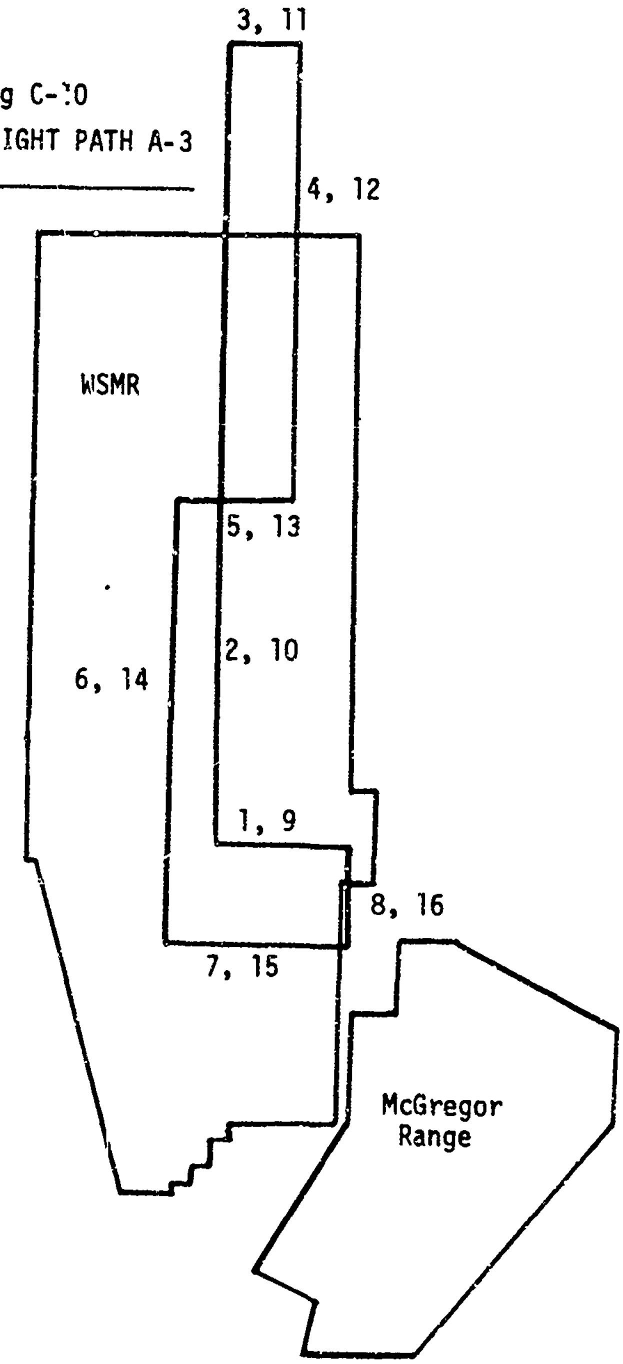
LEG NO	ALTITUDE (Feet)	MAGNETIC COURSE (Degrees)	END LATITUDE	END LONGITUDE	DISTANCE (NM)	TAS (Knots)	TIME (Min)	ELAPSED TIME (Hr+Min)
1/9	Low 500	257	32°52'N	106°24'W	15	250	05	0+05/1+08
2/10	1000	347	34°19'N	106°24'W	87	250	21	0+26/1+29
3/11	1000	077	34°19'N	106°13'W	8	250	02	0+28/1+31
4/12	1000	167	33°29'N	106°13'W	50	250	12	0+40/1+43
5/13	1000	257	33°29'N	106°31'W	13	250	03	0+43/1+46
6/14	1000	167	32°41'N	106°31'W	48	250	12	0+55/1+58
7/15	1000	077	32°41'N	106°06'W	20	250	05	1+00/2+03
8/16	1000	347	32°52'N	106°06'W	11	250	03	1+03/2+06

TOTAL DISTANCE 504 NM *

*Total distance reflects two complete circuits of the flight path.

Fig C-10

C-130 FLIGHT PATH A-3



1 in = 20 NM

0 10 20

TABLE C-XI

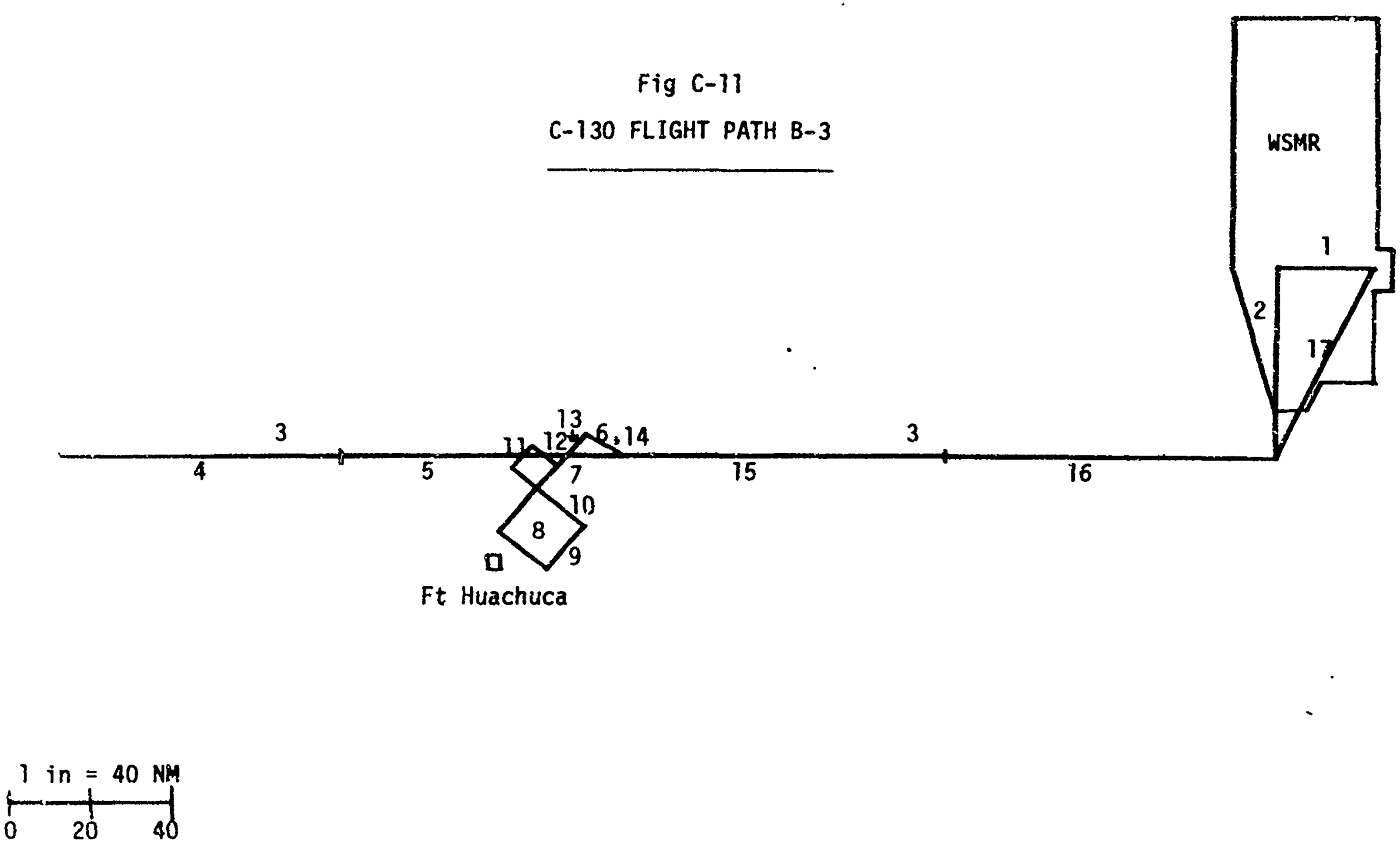
C-130 FLIGHT PATH B-3

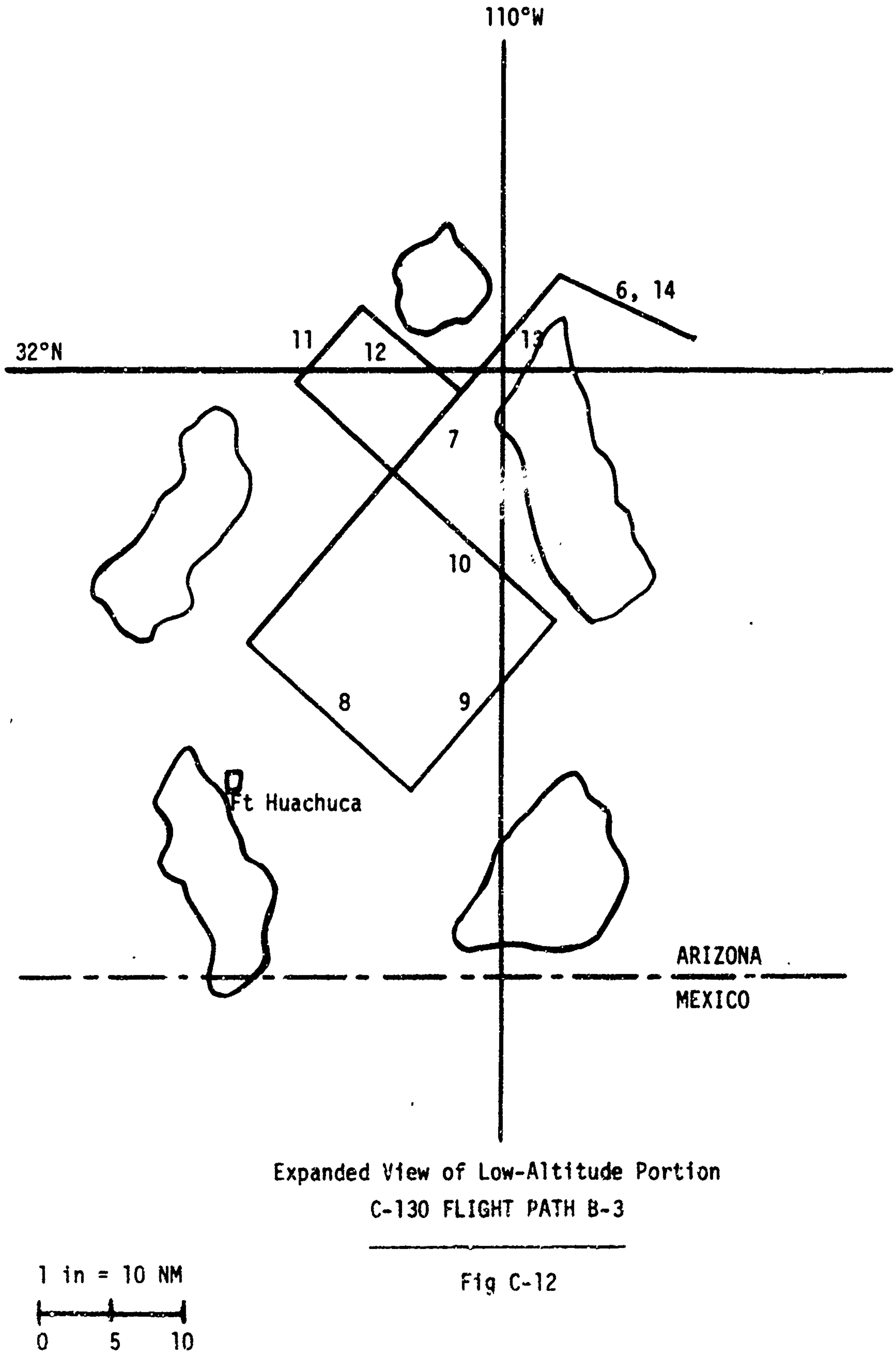
LEG NO	ALTITUDE	MAGNETIC COURSE (Degrees)	END LATITUDE	END LONGITUDE	DISTANCE (NM)	TAS (Knots)	TIME (Min)	ELAPSED TIME (Hr+Min)
1	Climb	257	32°52'N	106°35'W	25	190	08	0+08
2	Climb	167	32°02'N	106°35'W	50	190	16	0+24
3	30 K	257	32°02'N	112°29'W	300	300	60	1+24
4	30 K	077	32°02'N	111°07'W	69	300	14	1+38
5	Descent IP	077	32°02'N	109°45'W	70	275	15	1+53
6	Low 2000'	282	32°07'N	109°55'W	10	250	03	1+56
7	Low	208	31°42'N	110°21'W	33	250	08	2+04
8	Low	118	31°32'N	110°07'W	15	250	04	2+08
9	Low	028	31°43'N	109°56'W	15	250	04	2+12
10	Low	298	31°59'N	110°17'W	24	250	06	2+18
11	Low	028	32°04'N	110°11'W	7	250	02	2+20
12	Low	118	31°58'N	110°04'W	9	250	02	2+22
13	Low	028	32°07'N	109°55'W	11	250	03	2+25
14	Low	102	32°02'N	109°45'W	10	250	03	2+28
15	Climb	078	32°02'N	108°17'W	80	200	24	2+52
16	30 K	078	32°02'N	106°35'W	81	300	17	3+09
17	Descent	013	32°52'N	106°06'W	56	275	12	3+21

TOTAL DISTANCE 865 NM

Fig C-11

C-130 FLIGHT PATH B-3





Expanded View of Low-Altitude Portion
C-130 FLIGHT PATH B-3

Fig C-12

TABLE C-XII

C-130 FLIGHT PATH B-4

LEG NO	ALTITUDE	MAGNETIC COURSE (Degrees)	END LATITUDE	END LONGITUDE	DISTANCE (NM)	TAS (Knots)	TIME (Min)	ELAPSED TIME (Hr+Min)
1	Climb	347	35°07'N	106°06'W	135	220	37	0+37
2	30 K	318	28°57'N	108°39'W	260	300	52	1+29
3	Descent IP	230	38°34'N	109°37'W	50	275	11	1+40
4	Low	323	38°57'N	109°48'W	25	250	06	1+46
5	Low	238	38°50'N	110°17'W	23	250	06	1+52
6	Low	169	37°06'N	110°25'W	103	250	25	2+17
7	Low	076	37°06'N	110°13'W	10	250	03	2+20
8	Low	346	37°16'N	110°13'W	10	250	03	2+23
9	Low	265	37°16'N	110°24'W	10	250	03	2+26
10	Low	349	38°50'N	110°17'W	93	250	22	2+48
11	Low	058	38°57'N	109°48'W	23	250	06	2+54
12	Low	143	38°34'N	109°37'W	25	250	06	3+00
13	Climb	050	38°57'N	108°39'W	50	200	15	3+15
14	Climb-30 K	138	35°07'N	106°06'W	260	300	52	4+07
15	Descent	167	32°52'N	106°06'W	135	290	28	4+35

TOTAL DISTANCE 1212 NM

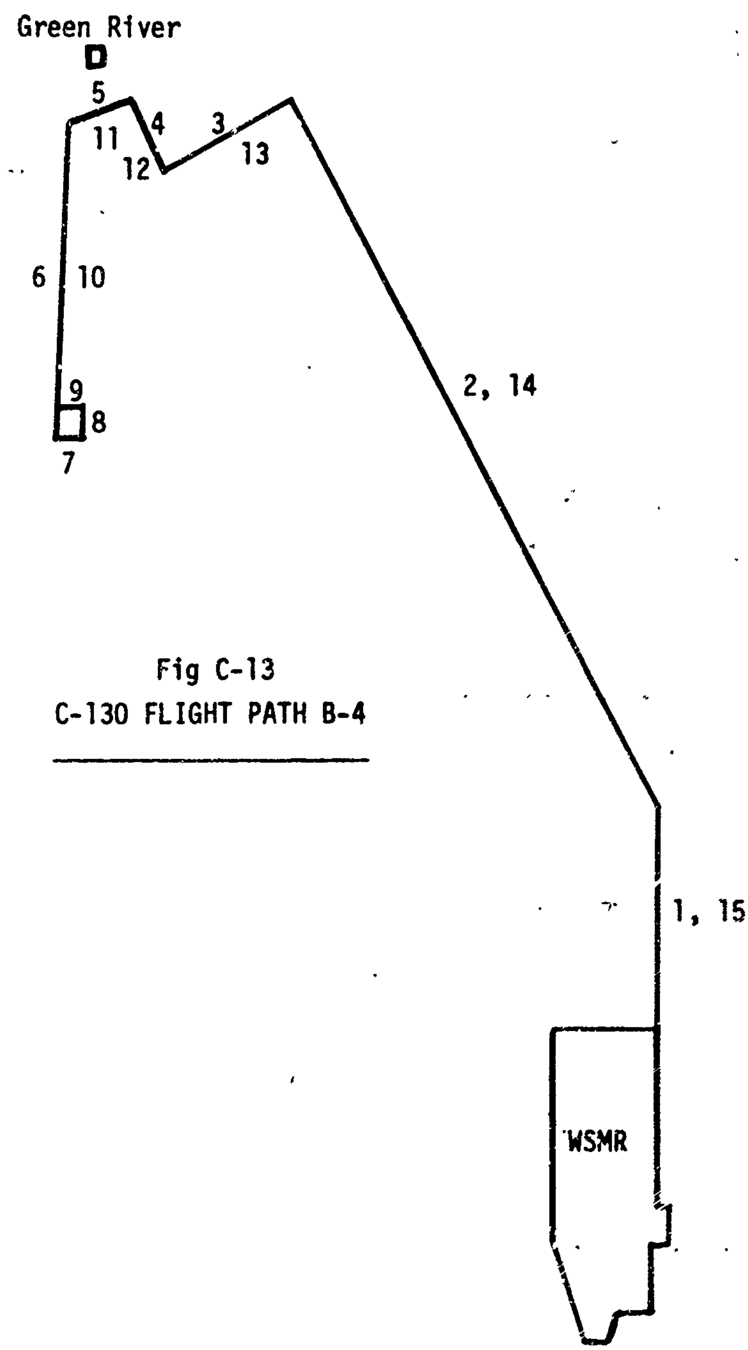


Fig C-13
C-130 FLIGHT PATH B-4

1 in - 60 NM
0 30 60

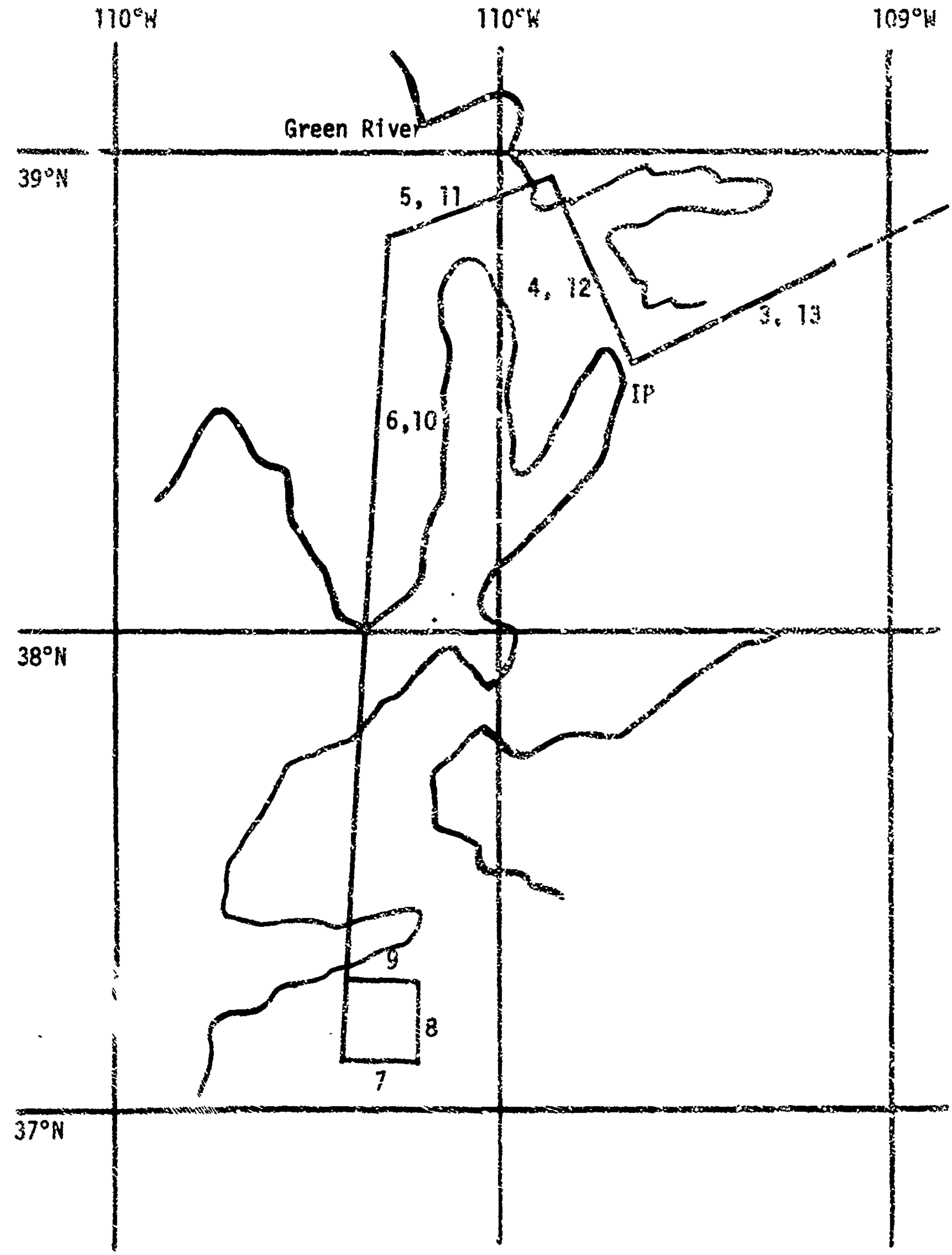


Fig C-14

Expanded View of Low-Altitude Portion

C-130 FLIGHT PATH B-4

1 in = 20 NM

0 10 20

APPENDIX D

INSTRUMENTATION

This appendix should be considered as a guide only. Each program has its own peculiarities and must be treated individually.

1. EQUIPMENT FURNISHED BY CIGTF

1.1 Ground

1.1.1 Oscillograph recorders.

1.1.2 Tape playback capability.

1.1.3 FM - Ground Station.

1.1.4 Miscellaneous test equipment.

1.2 Airborne

1.2.1 Magnetic Tape Recorder. Fourteen channels with FM or direct capability. (Table D-I shows a typical tape recorder track assignment scheme.)

1.2.2 VCO Complexes. Up to four six-packs with calibrators. (Table D-II lists a typical VCO channel assignment scheme.)

1.2.3 Visicorders, 24 channel, 1108-2-04-700HK; Galvo amplifiers, TG6A-500.

1.2.4 Signal Conditioner. For aircraft 400 cps and 28 vdc power, altitude, temperature, air flow and any last minute system signals required on a limited basis.

1.2.5 Vibration transducers and amplifiers.

1.2.6 Timing generator.

2. CUSTOMER SUPPORT

The customer will be responsible for insuring that all system signals to be recorded (except as noted in paragraph 1.2.4, above) are properly conditioned prior to system delivery. Paragraph 3 defines the constraints imposed by CIGTF equipment.

3. SYSTEM SIGNAL PARAMETERS

3.1 Tape Recorder

3.1.1 Voltage Levels. Zero to +5 volts or ±2.5 volts. Single-ended referenced to aircraft ground.

3.1.2 Frequency. For FM, 1.25 kc maximum; for Direct, 10 kc maximum.

3.1.3 Format. Digital outputs will utilize a Manchester type code 0 to 5 volts. For Direct recording the pulse train will not be gated and bit rate will be 1 kc to 10 kc. For FM recording bit rate will not exceed 1 kc and a gated pulse train is permissible.

3.2 Visicorder

3.2.1 Voltage Levels. Zero to $\pm$5v.

3.2.2 Maximum System Signal Output Impedance. 10 k ohms

3.2.3 Input Impedance into Recording Amplifier. 47 k ohms in parallel with 300 picofarads.

3.2.4 Maximum Frequency. 4.8 kc

4. ALTITUDE TRANSDUCER

Altitude transducers available for use are Wallace O. Leonard Model No. 503654-39. These are analog devices. Details include:

4.1 Altitude. Zero to 80,000 ft

4.2 Resistance. Zero to 5 k ohms, or 6.25 ohms/100 ft pressure altitude.

4.3 Maximum Voltage Input. 75 vdc or ac (rms)

5. THEODOLITES

Wild T-3s

TABLE D-I

TYPICAL TAPE RECORDER TRACK ASSIGNMENT

Track	Type	Function
1	Direct	Voice Annotation
2	FM	Spare
3	Direct	A Link
4	FM	Spare
5	Direct	B Link
6	FM	X Vibration
7	Direct	12.5 kc Reference
8	Direct	12.5 kc Reference
9	Direct	C Link
10	FM	Z Vibration
11	Direct	Position Data
12	FM	Spare
13	Direct	Frame Mark
14	FM	Y Vibration

TABLE D-II

TYPICAL VCO CHANNEL ASSIGNMENT

VCO (kc)	Information Bandwidth (cps)	A Link	B Link	C Link
10.5	160	X-gyro Torquer ±2.5v	Y-gyro Torquer ±2.5v	Z-gyro Torquer ±2.5v
7.35	110	X Accelerometer ±2.5v	Y Accelerometer ±2.5v	S Slow Presence 0-5v (threshold)
5.4	81	Alt Limit 1 0-5v	Alt Limit 2 0-5v	Excess Brightness 0-5v
3.9	59	Cloud Detector 0-5v	Doppler "On" 0-5v	Spare Temp 0-5v
3.0	45	Cooling Air Temp 0-5v	Cabin Air Temp 0-5v	SIR U Temp 0-5v
2.3	35	Computer 400 Amp 0-5v	Aircraft 400 Amp 0-5v	Low Line Voltage 0-5v
1.7	25	Overtemp No-Go 0-5v	Doppler Memory 0-5v	Camera Release 0-5v
Direct		Computer 400 Freq		Aircraft 400 Freq

APPENDIX E

SYSTEM CALIBRATION PROCEDURES

1. INTRODUCTION

System calibration serves two purposes:

(1) It determines the component parameters (accelerometer bias and scale factor, gyro compensable drift) which are inserted into the system to offset the known systematic errors.

(2) It provides a quantitative knowledge of system initial conditions (component parameters and misalignments) which may be used as *a priori* inputs to the computer program used to isolate and identify significant system error sources. For this reason, system calibration procedures will be developed by the CIGTF project analyst. In most cases, the contractor's recommended procedure may be used as a guide to introduce the analyst to the basic system operation.

2. SPECIFIC SYSTEM CALIBRATION PROCEDURES (Sample)

The procedures outlined below are based on a specific system. They are included here as an example of the procedures to be used.

2.1 The system will be calibrated while mounted on a gyro test table in the orientation shown in Figure E-1 (position No. 1 in Table E-I). The table top can be positioned accurately to within 2 seconds of arc of any selected heading. Additionally, the top may be tilted about the E-W axis to a geodetic vertical accuracy of approximately 4 arc seconds. Once the system has been mounted on the table, table positions will be used as reference. Six calibration positions which denote axis orientation and reference acceleration will be used, as shown in Table E-I.

POSITION NO.	AXIS ORIENTATION			REFERENCE ACCELERATION ($\bar{a}_I$)		
	X	Y	Z	a_{I_x}	a_{I_x}	a_{I_x}
1	N	W	U	0	0	g
2	W	S	U	0	0	g
3	W	D	S	0	-g	0
4	D	E	S	-g	0	0
5	E	U	S	0	g	0
6	U	W	S	g	0	0

TABLE E-1

The table axes will be positioned in turn to each of the positions in Table E-I. The outputs from the x and y velocity meters will be recorded for a specified period of time. These outputs will be used in a least squares program to determine the misalignment, velocity meter bias and scale factor.

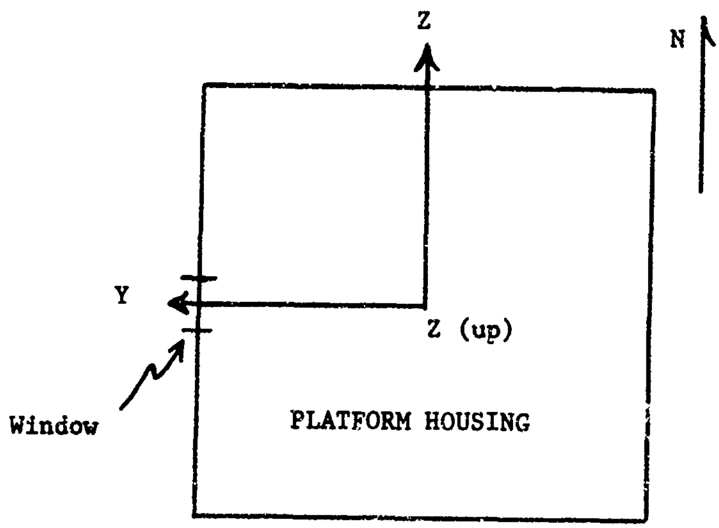


FIGURE E-1

2.2 **Axis Misalignment, Velocity Meter Bias and Scale Factor.** The velocity meters are shimmed 1.5° (±10 arc seconds) from the x and y axes. In addition, there is probably an additional slight misalignment of all three axes (≤30 arc minutes). These misalignments are shown in Figure E-2.

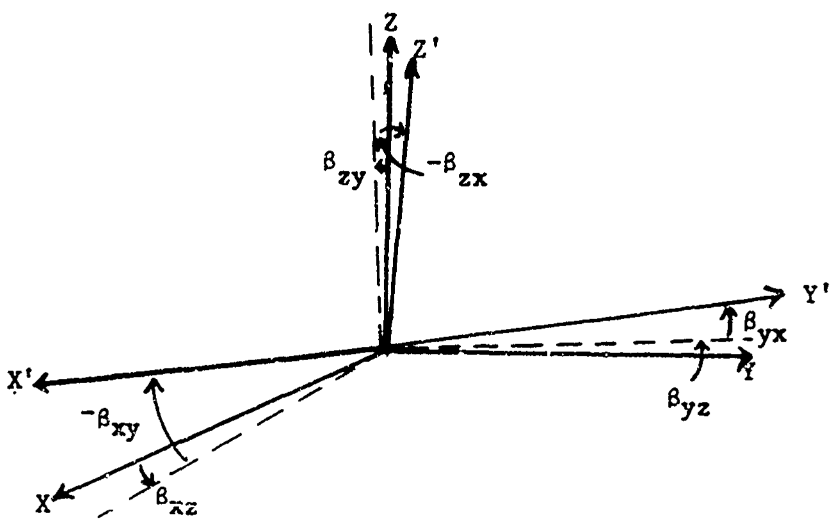


FIGURE E-2

The matrix, $\bar{\bar{M}}$, relating the actual and reference positions of the axis set is:

$$\bar{\bar{M}} = \begin{bmatrix} \cos\beta_{xy}\cos\beta_{xz} & \beta_{xz} & -\sin\beta_{xy} \\ -\beta_{yz} & \cos\beta_{yx}\cos\beta_{yz} & \sin\beta_{yx} \\ \beta_{zy} & -\beta_{zx} & \cos\beta_{zx}\cos\beta_{zy} \end{bmatrix}$$

where the notation β_{AB} signifies the angle of misalignment of A due to a rotation about B. In the derivation of this matrix, small angle assumptions were made for all angles except β_{xy} and β_{yx}. *

Since this is the matrix which represents axis misalignment, it also may be used to relate reference input acceleration to sensed acceleration. If there were no misalignment, each of the sensitive axes would detect reference input acceleration along that axis exactly. However, in the real case

$$\bar{a}_S = \bar{\bar{M}}\,\bar{a}_I$$

where $\bar{a}_S$ = sensed acceleration

$\bar{a}_I$ = reference input acceleration.

The null bias, $\bar{K}_o$, and the scale factor, $\bar{\bar{K}}_1$, are related to the output by the expression:

$$\overline{\text{Output}} = \bar{K}_o + \bar{\bar{K}}_1\bar{a}_S = \bar{K}_o + \bar{\bar{K}}_1\bar{\bar{M}}\,\bar{a}_I$$

$$= \begin{bmatrix} K_{ox} \\ K_{oy} \\ K_{oz} \end{bmatrix} + \begin{bmatrix} K_{1_x} & 0 & 0 \\ 0 & K_{1_y} & 0 \\ 0 & 0 & K_{1_z} \end{bmatrix} \begin{bmatrix} \bar{\bar{M}} \end{bmatrix} \begin{bmatrix} a_{I_x} \\ a_{I_y} \\ a_{I_z} \end{bmatrix}$$

where K_{oi} = bias of the i^{th} accelerometer

K_{1i} = scale factor of the i^{th} accelerometer

a_{I_j} = reference input acceleration on j^{th} coordinate axis.

---

* For an angle of 30 min, the sine of the angle differs from the radian measure of the angle by only one part in 10^7. The cosine, however, differs from one by one part in 25,000, so the cosine of the angle is retained for accuracy. In addition, for maximum angles of 1.5° (β_{xy} and β_{yx}), the error involved in assuming that the order of rotation is immaterial is three parts in 10,000.

A least squares solution for the six table positions is defined as follows:

$$\begin{bmatrix} \\ \\ \bar{\bar{N}} \\ \\ \\ \end{bmatrix} \begin{bmatrix} \\ \\ COEF_j \\ \\ \\ \end{bmatrix} = \begin{bmatrix} \sum\limits_i out_j(i) \\ \sum\limits_i out_j(i)\cdot a_{I_x}(i) \\ \sum\limits_i out_j(i)\cdot a_{I_y}(i) \\ \sum\limits_i out_j(i)\cdot a_{I_z}(i) \end{bmatrix}$$

where $$\bar{\bar{N}} = \begin{bmatrix} \sum\limits_i 1 & \sum\limits_i a_{I_x}(i) & \sum\limits_i a_{I_y}(i) & \sum\limits_i a_{I_z}(i) \\ \sum\limits_i a_{I_x}(i) & \sum\limits_i a^2_{I_x}(i) & \sum\limits_i a_{I_x}(i)a_{I_y}(i) & \sum\limits_i a_{I_x}(i)a_{I_z}(i) \\ \sum\limits_i a_{I_y} & \sum\limits_i a_{I_y}(i)a_{I_x}(i) & \sum\limits_i a^2_{I_y}(i) & \sum\limits_i a_{I_y}(i)a_{I_z}(i) \\ \sum\limits_i a_{I_z} & \sum\limits_i a_{I_z}(i)a_{I_x}(i) & \sum\limits_i a_{I_z}(i)a_{I_y}(i) & \sum\limits_i a^2_{I_z}(i) \end{bmatrix}$$

i = index of calibration position (range 1-6)

j = accelerometer (x or y)

K = reference axis (x, y, or z)

a_{I_k} = acceleration along k^{th} axis in i^{th} calibration position

$out_j(i)$ = output of j^{th} accelerometer in i^{th} position.

For the six calibration positions,

$$\bar{N} = \begin{bmatrix} 6 & 0 & 0 & 2g \\ 0 & 2g^2 & 0 & 0 \\ 0 & 0 & 2g^2 & 0 \\ 2g & 0 & 0 & 2g^2 \end{bmatrix},$$

$$COEF_x = \begin{bmatrix} K_{ox} \\ K_{1x} \cos \beta_{xy} \cos \beta_{xz} \\ K_{1x} \beta_{xz} \\ -K_{1x} \sin \beta_{xy} \end{bmatrix},$$

and

$$COEF_y = \begin{bmatrix} K_{oy} \\ -K_{1y} \beta_{yz} \\ K_{1y} \cos \beta_{yx} \cos \beta_{yz} \\ K_{1y} \sin \beta_{yx} \end{bmatrix}$$

where K_{oj} = bias of j^{th} velocity meter

K_{ij} = scale factor of j^{th} velocity meter

β_{jk} = misalignment of j^{th} velocity meter about k^{th} axis.

The the solutions for $COEF_j$ can be reduced to obtain the desired misalignments, bias and scale factor. For example, $COEF_x$ becomes:

$$K'_{1x} = K_{1x} \cos \beta_{xy} \cos \beta_{xz}$$

$$\beta'_{xz} = \frac{K_{1x}\,\beta_{xz}}{k'_{1x}}$$

$$\beta'_{xy} = \sin^{-1}\left(\frac{-K_{1x} \sin \beta_{xy}}{K'_{1x}}\right).$$

Then define $$\gamma = \cos \beta'_{xz} \cos \beta'_{xy}$$

and redefine $$K_{1x} = \frac{K'_{1x}}{\gamma} \quad \text{(output/g)}$$

$$\beta_{xz} = \beta'_{xz} \cdot \gamma \quad \text{(radians)}$$

$$\beta_{xy} = \beta'_{xy} \cdot \gamma \quad \text{(radians)}$$

$$K_{ox} = \frac{K_{ox}}{K_{1x}} \quad (\mu g)$$

which are the bias, scale factor and misalignment angles of the x velocity meter.

As part of the calibration results, we desire the set of

$$C_{ij} = \frac{\overline{P}_i \cdot \overline{COEF}_j - out_j(i)}{K_{1j}}$$

which constitutes the residuals of the least squares analysis.

2.3 Gyro Calibration Procedure

2.3.1 Gimbal cage the platform to the position shown in Figure E-1. Start the gyros and allow the platform to stabilize. Level the platform using the velocity meters, and hold the azimuth by sighting on the mirror with the autocollimator and continuously correcting the z gyro pulses.

2.3.2 Using the VM leveling loops, count the controlling pulses to the x and y gyros for a prescribed period of time (~3 minutes). The z zero pulses in the autocollimator azimuth loop should also be counted for the same prescribed period.

2.3.3 Torque the z gyro at a constant rate to rotate the platform -90° about azimuth. This will require about one and a half hours. Rotate the platform housing until it is positioned so that the mirror may be seen by the north theodolite as in Figure E-3.

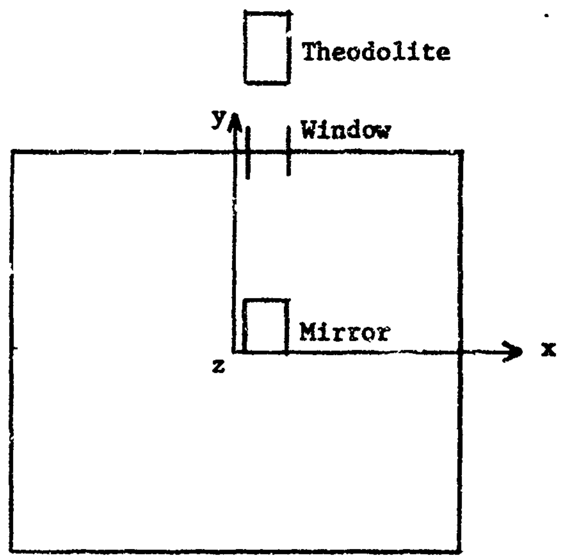


FIGURE E-3

2.3.4 Stop the platform torquing when the rotation attains -90° as determined by the theodolite. Count the torquing pulses and time required for rotation. Once again, use the VM leveling loops to count the controlling pulses to the x and y gyros.

2.4 Gyro Calibration Equations

2.4.1 A two-position calibration is performed on the level gyros, with the input axis aligned north and east. Gyro scale factor and bias are found by solving these equations:

$$\text{BIAS} = \dot{P}_E$$

$$\text{SF} = \frac{\Omega_N}{\dot{P}_N - \dot{P}_E}$$

where $\dot{P}_E$ = mean pulse rate with input axis east

$\dot{P}_N$ = mean pulse rate with input axis north

Ω_N = north component of earth rate.

2.4.2 The azimuth gyro is calibrated by accumulating data first at a zero degree heading, then while azimuth is torqued through an angle of 90°. Equations used are:

$$\text{BIAS} = \frac{\dot{P}_1(\omega - \Omega_Z) + \Omega_Z \dot{P}_2}{\omega}$$

$$\text{SF} = \frac{\omega}{\dot{P}_1 - \dot{P}_2}$$

where $\dot{P}_1$ = mean pulse rate at zero degrees heading

$\dot{P}_2$ = mean pulse rate while torquing in azimuth

ω = mean torquing rate

Ω_Z = vertical component of earth rate.

2.4.3 A two position (up, down) calibration is used to find velocity meter scale factor and bias. The following equations are used:

$$\text{BIAS} = \frac{\dot{P}_D + \dot{P}_U}{2}$$

$$\text{SF} = \frac{\dot{P}_U - \dot{P}_D}{2g}$$

where $\dot{P}_U$ = mean velocity pulse rate with sensitive axis up

$\dot{P}_D$ = mean velocity pulse rate with sensitive axis down

g = local gravity (32.124 ft/sec^2).

APPENDIX F

PRE-FLIGHT AND POST-FLIGHT PROCEDURES

1. C-130 SYSTEM TEST PRE-FLIGHT

1.1 The C-130 aircraft will be positioned with its heading aligned as close to True North as possible in a predetermined parking and run-up area. Aircraft compass heading plus local variation will be used to determine the best available true heading (BATH). Using this procedure, obtainable heading accuracy is expected to be ± one degree. A reference mark indicating nose wheel position in the parking area will be painted on the ramp and the exact coordinates of this mark will be determined by survey. An accurate initial position will thus be available for system initial condition insert.

1.2 An auxiliary ground power unit will normally be used to provide both ac and dc power for the system(s) and test instrumentation during warm-up and alignment. The tape recorder(s) and visicorder(s) will be checked to insure proper operation and complete recording capability. The timing signal generator will be placed on board the aircraft, integrated with the test instrumentation and checked for accuracy against the IRIG-B timing signals transmitted by WSMR. Proper operation of all other instrumentation and communications will be ascertained prior to engine start.

1.3 At the completion of system alignment, the inner gimbal misalignment from True North will be determined using the following procedure:

One theodolite will be mounted on the appropriate stand immediately outside the C-130 cargo door or F-106 nose bay. An azimuth reading within five arc seconds will be made on the inner gimbal mirror. The theodolite will be swung and aligned with a second reference (bench mark) and the first theodolite (see Figure F-1). These measurements will accurately determine, within five arc seconds, the true heading of the azimuth gimbal. In this manner, the azimuth misalignment of the IMU can be determined prior to flight.

1.4 The entire pre-flight procedure will require approximately 30 minutes, plus the system warm-up and alignment time.

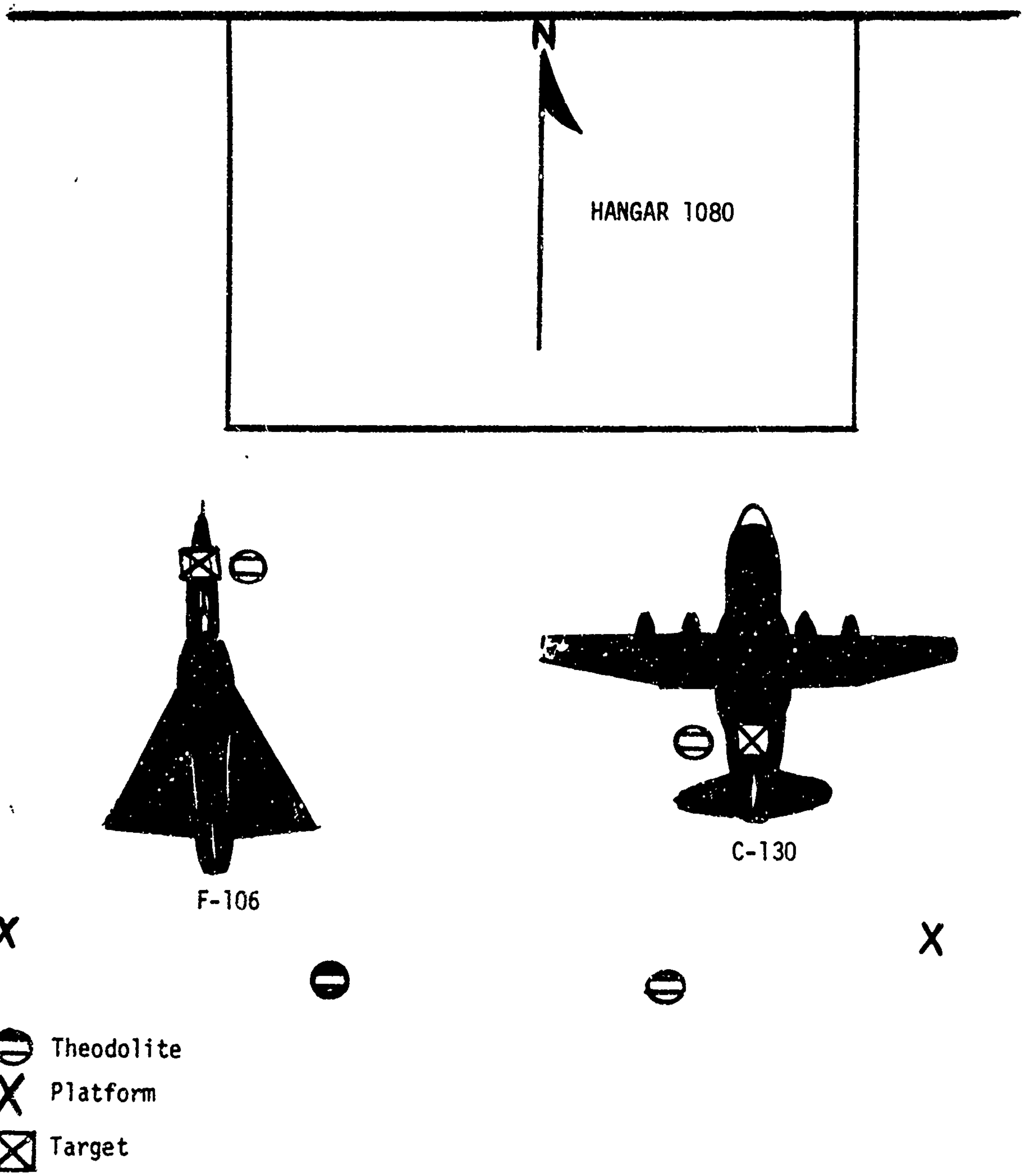
1.5 The system tape recorder(s) and visicorder(s) must be in operation from approximately five minutes prior to switching to navigate until approximately ten minutes after terminal parking.

2. C-130 SYSTEM TEST POST-FLIGHT

2.1 The aircraft will be landed on the available runway that minimizes the taxi time to the calibrated reference mark. The aircraft will be taxied over the pre-taxi marks as closely as possible to a true northerly heading using the aircraft gyro heading indicator. Power will be switched to an auxiliary ground power unit and the main engines shut down. The inner gimbal misalignment will be determined as before and compared to the original reading to obtain differential azimuth gyro drift.

Fig F-1

OPTICAL ALIGNMENT LAYOUT



2.2 The test (flight) will normally be considered complete when the aircraft has returned to its parking position, the aircraft engines have been shut down, and the inner gimbal misalignment measured. The magnetic tape recorder will normally be shut off immediately after engine shutdown.

2.3 System(s) will not normally be operated longer than 15 minutes after aircraft engine shutdown, although at the discretion of the CIGTF project personnel, the system may be operated for any additional period desired.

3. F-106 & HELICOPTER SYSTEM TEST PRE-FLIGHT AND POST-FLIGHT

F-106 and helicopter procedures will be similar to those in the C-130 with the following exception:

When the tape recorder has been started, a burst of IRIG-B time will be recorded from a portable timing generator or land line. This will be used as a reference start. A precision 12.5 kc reference oscillator will be carried on board the aircraft to provide a continuous reference time base. At the completion of the flight, the IRIG-B portable timing generator signal will again be recorded to provide a reference stop time signal.

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13. ABSTRACT

The designation of the Central Inertial Guidance Test Facility (CIGTF) as the DoD focal point for aircraft inertial navigator test and evaluation required that a generalized test plan be written to govern all future tests. This document outlines such a Standardized Test, including test philosophy and objectives, the test approach and an outline of the test procedure. It provides the reader with an understanding of the CIGTF aircraft inertial navigator test capabilities, the types of test programs currently available, and the requirements necessary for an agency to enter systems in these programs. Six appendices, which cover areas such as analysis methods, laboratory testing, instrumentation, are included to provide the project test engineer with additional detailed information.

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